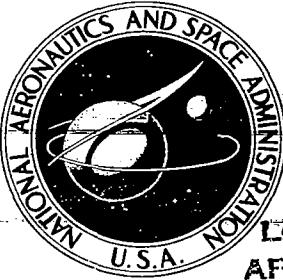


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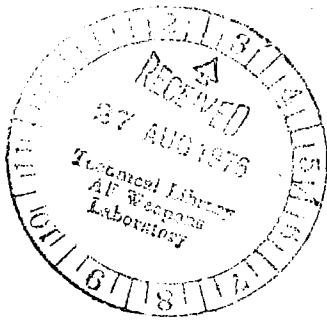
**STUDY OF A FAIL-SAFE ABORT SYSTEM  
FOR AN ACTIVELY COOLED  
HYPERSONIC AIRCRAFT****Volume I - Technical Summary***C. J. Pirrello and R. L. Herring**Prepared by*

MCDONNELL AIRCRAFT COMPANY

St. Louis, Mo. 63166

for Langley Research Center

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## FOREWORD

This is one of two report volumes describing the technical results of the study of a fail-safe abort system for an actively cooled hypersonic aircraft. The other report, detailing the analyses, is "Volume II, Technical Report".

The overall objective of this study was to conceptually design and evaluate a fail-safe active cooling system which will be used in an abort mode from cruise Mach numbers of 3 to 6. The study was conducted in accordance with the requirements and instructions of NASA RFP 1-15-4807 and McDonnell Technical Proposal, Report MDC A2961, with minor revisions mutually agreed upon by NASA and MCAIR. The study was conducted using customary units for the principal measurements and calculations. Results were converted to the International System of Units (S.I.) for the final reports.

Mr. Charles J. Pirrello was the MCAIR Study Manager with Mr. Ralph L. Herring as Principal Investigator. The study was conducted within MCAIR Advanced Engineering which is managed by Mr. Harold D. Altis, Director, Advanced Engineering Division. The study team was an element of Advanced Concepts, supervised by Mr. Dwight H. Bennett.

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LIST OF ABBREVIATIONS AND SYMBOLS

Abbreviations                           Definition

ACS	Active Cooling System
AL	Aluminum
FDS	Failure Detection System
FS	Fuselage Station
H-X	Heat Exchanger
INS	Insulation
IR	Infrared
NM	Nautical Miles
PCM	Phase Change Material
Wt	Weight

Symbols

A	Area
Btu	British thermal unit
CL	Centerline
D	Diameter
°F	Degrees Fahrenheit
ft	Feet
g	Acceleration due to gravity
h	Honeycomb depth
h'	Distance between centroids of outer and inner skin
hr	hour
in	Inch
lbf	Pounds force
lbm	Pounds mass
L/D	Lift to Drag Ratio
LH <sub>2</sub>	Liquid Hydrogen
min	Minute
$\dot{m}$	Mass flow rate

LIST OF ABBREVIATIONS AND SYMBOLS (CONT'D)

Abbreviations

Definition

$n_z$	Limit load factor in vertical plane (positive up)
P	Pitch, Pressure
psid	Pounds force per square inch, differential
q	Dynamic pressure
$\dot{q}$	Heating rate per unit area
Q	Integrated heat per unit area = $\int \dot{q} dt$
$\dot{Q}$	Integrated heating rate = $\int \dot{q} dA$
S	Reference lifting area
sec.	Second
t	Thickness
T	Temperature
W	Mass
W/S	Wing loading
X	Distance, thickness

Greek Symbols

$\alpha$	Angle of attack
$\Delta$	Difference
$\tau$	Descent time
$\phi$	Bank angle

SI Units

g	Gram (mass)
J	Joule (heat)
K	Kelvin (temperature)
m	Meter (length)
N	Newton (force)
Pa	Pascal (pressure)
W	Watt (power)
s	Second (time)

LIST OF ABBREVIATIONS AND SYMBOLS (CONT'D)

SI Prefixes

m	Milli ( $10^{-3}$ )
c	Centi ( $10^{-2}$ )
k	Kilo ( $10^3$ )
M	Mega ( $10^6$ )

Subscripts

avg	Average
c	Coolant
f	Fuel
i	Inner skin
ins	Insulation
L	Lower
min	Minimum
max	Maximum
o	Outer skin
s	Skin
sil	Silicone
t	Tube
T	Total
U	Upper
w	Wetted, wall
X	Aircraft axis - longitudinal
Z	Aircraft axis - vertical

x

STUDY OF A FAIL-SAFE ABORT SYSTEM FOR AN ACTIVELY COOLED  
HYPERSONIC AIRCRAFT, VOLUME I, TECHNICAL SUMMARY  
by C. J. Pirrello and R. L. Herring  
McDonnell Aircraft Company

1. SUMMARY

A detailed study was conducted to conceptually design and evaluate a fail-safe actively cooled structural system which will be used in an abort mode from cruise Mach numbers of 3 to 6.

The specific objectives of this study were:

- o To determine and evaluate means of failure detection of those active cooling systems failures requiring abort.
- o To optimize abort mode descent trajectories for cruise Mach numbers of 3 to 6 for minimum heat load.
- o To define and evaluate thermostructural concepts for actively cooled structure which will minimize both weight and maximum structural temperature during the abort.

The baseline aircraft configuration used throughout the study was a Mach 6 actively cooled, liquid hydrogen fueled, transport (Reference (2)) with a cooled surface area of  $2980 \text{ m}^2$  ( $32,134 \text{ ft}^2$ ). Figure 1 illustrates the aircraft.

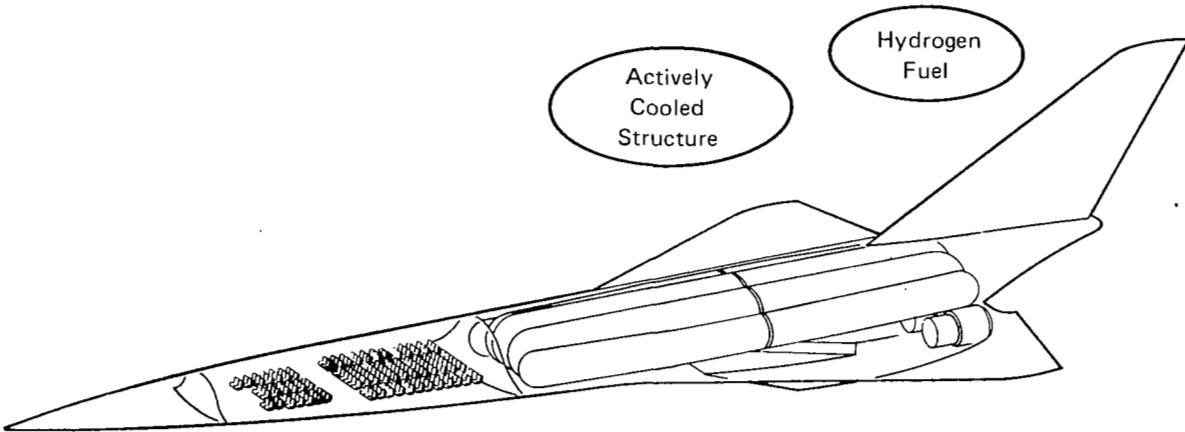


FIGURE 1  
BASELINE AIRCRAFT CONFIGURATION

The Mach 6 design was used for Mach 3 and 4.5 studies in addition to Mach 6. This aircraft is far from optimum at the reduced speeds. However, use of this configuration throughout the study range of flight speeds served its purpose. It provided a large hypersonic baseline aircraft design with known characteristics which were compatible with study objectives. The aircraft was held to a fixed size for all study Mach numbers. Thus, the results obtained are not mission, or range dependent.

A variety of trade studies were conducted in the areas of cooling system/actively cooled structure failure detection, abort descents from cruise, and thermostructural design in order to evaluate the fail-safe systems.

It was determined that overall operation of the basic active cooling system can be monitored for failure by conventional instrumentation. The complicating factor in detection of a failure is the large surface area of the aircraft. A promising concept for sensing a high out-of-tolerance skin temperature is identified.

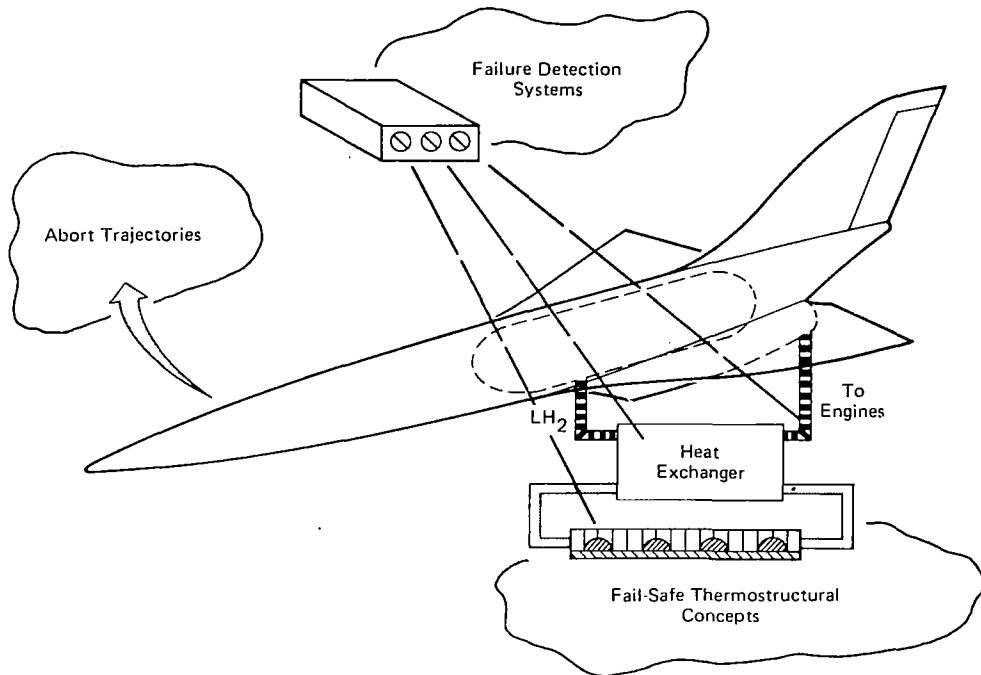
The low heat load descent trajectories selected for abort from cruise Mach number are all maximum g-load pull-up maneuvers with transition to a high lift coefficient descent. Aircraft angle of attack was limited to 20 degrees. These descent trajectories were found to be very effective in the reduction of aircraft total heat load. In general, the descent heat load ranged from 32 to 25 percent of normal maximum lift-to-drag ratio descent heat load for start of abort Mach numbers of 3 and 6, respectively.

A large number of potential candidate thermostructural concepts were screened and evaluated for possible application to aircraft designed for operation within the Mach 3 to 6 flight regime. The thermostructural concepts finally selected for Mach 3, 4.5 and 6 cruise were designed based on both cruise and abort considerations and provide adequate abort capability.

Three specific point design "Fail-Safe Systems" were conceptually developed. Figure 2 summarizes the weights for these systems.

An extremely important consideration was the ability of the thermostructural concepts to restrict the absorbed heat to a level equal to, or less than, the heat capacity of the hydrogen fuel flow. All three conceptual "Fail-Safe" system designs met this requirement as shown in Figure 3.

The hydrogen fuel heat sink capacity illustrated in Figure 3 was derived from typical characteristics of duct burning turbofan engines at Mach 3, and for the Mach 4.5 and 6 cases, turboramjet engines.



Fail-Safe System Weight - Mg(lbm)	Cruise Mach Number		
	3	4.5	6
Actively Cooled Structure	37.7 (83,227)	37.7 (83,227)	37.7 (83,227)
Thermal Protection System	1.5 ( 3,093)	5.2 (11,252)	8.4 (18,476)
Active Cooling System	2.3 ( 4,974)	2.6 ( 5,645)	3.2 ( 6,955)
Failure Detection System	0.5 ( 1,081)	0.5 ( 1,212)	0.6 ( 1,343)
Total	42.0 (92,375)	46.0 (101,338)	49.9 (110,001)

**FIGURE 2  
FAIL-SAFE SYSTEM WEIGHTS**

Cruise Mach No.	Fail-Safe System Point Designs		
	3	4.5	6
Cruise Altitude km (ft)	22.9 (75,000)	28.3 (93,000)	32.0 (105,000)
Cooling System Heat Load (Start of cruise) MW (Btu/sec)	15.4 ( $1.46 \times 10^4$ )	18.1 ( $1.72 \times 10^4$ )	25.2 ( $2.39 \times 10^4$ )
Fuel Flow Heat Sink Capacity (Cruise Average) MW (Btu/sec)	15.4 ( $1.46 \times 10^4$ )	44.4 ( $4.21 \times 10^4$ )	54.2 ( $5.14 \times 10^4$ )
Heat Load-to-Heat Sink Ratio	1.00	0.41	0.46

**FIGURE 3  
FAIL-SAFE SYSTEMS HEAT SINK REQUIREMENTS ARE LOW**



## 2. INTRODUCTION

Previous studies of hydrogen fueled, high speed transport aircraft which utilized actively cooled structure showed the concept has potential advantages over hot structures. The use of hydrogen for fuel provides a heat sink source for the active cooling system to use in reducing aircraft skin and structural temperatures. The reduced temperature allows the use of conventional low temperature material which may provide a longer useful life and a reduced cost compared to some of the more exotic high temperature materials.

However, there are still numerous problems which require investigation before an optimum, workable, safe aircraft system is designed. Redundant systems are heavy and may not provide the best overall approach to safe aircraft. In addition, there may be failures which cannot be negated by redundant systems. Reference (3) indicated that with highly efficient aerodynamic and propulsion system approaches, the normal cruise fuel flow would not be sufficient to cool the entire aircraft. Thus, either an additional expendable heat sink source would be required or the heat load to the cooling system would have to be reduced. These are important issues, however, the more compelling issue is insuring the safety of the aircraft in all possible situations.

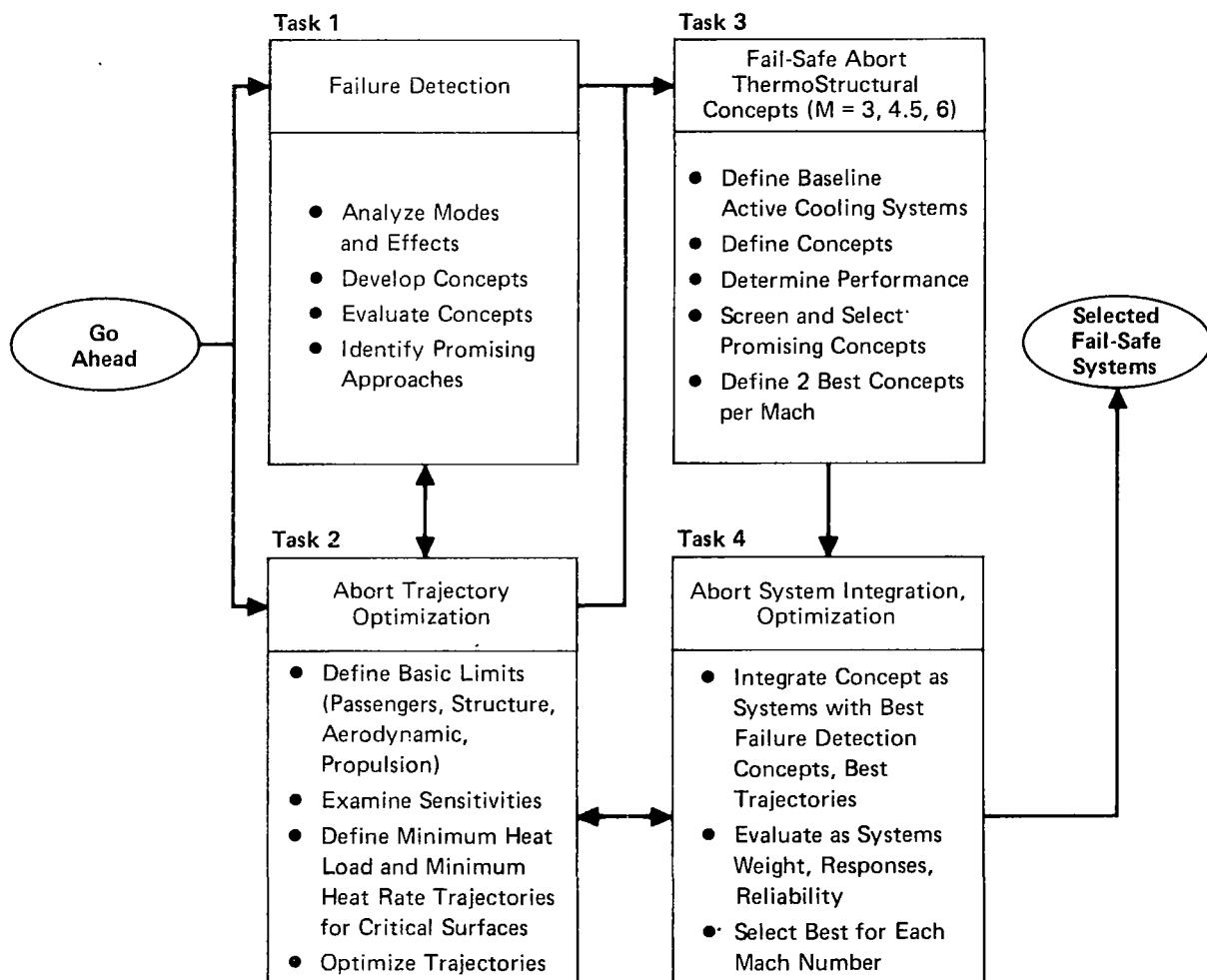
Reference (4) presented a fail-safe system concept as an alternative to a redundant active cooling system. This concept consisted of an abort maneuver by the aircraft and a passive thermal protection system (TPS) in the form of overcoat material for the aircraft skin. The abort maneuver provides a low-heat-load descent from normal cruise speed to a lower speed at which cooling is unnecessary, and the passive TPS allows the aircraft structure to absorb the abort heat load without exceeding critical structural temperatures. In addition, the passive TPS may solve the fuel flow problem, a consequence which would be most welcome.

On the basis of preliminary results obtained during conduct of Reference (4), it appeared that the fail-safe system concept warranted further consideration. Thus, this investigation laid the foundation for the study summarized herein.

The overall study emphasis was placed on the conceptual design and evaluation of fail-safe systems designed for use in abort modes from cruise Mach numbers of 3 to 6. The fail-safe concept depends on three basic factors; a

reliable method of detecting a failure or malfunction in the active cooling system or cooled structure, the optimization of abort trajectories which minimize the heat load to the aircraft, and fail-safe thermostructural concepts to minimize both the weight and the maximum temperature the structure will reach during descent. Thus, studies were conducted in these specific areas.

The overall study logic is illustrated by Figure 4. Each of the first three tasks, while distinctively different elements, involved an appreciable amount of interaction. The overall integration of the three elements evaluated all the pertinent interactions in the final evaluation and selection process.



**FIGURE 4**  
**FAIL-SAFE ABORT SYSTEM STUDY PLAN**

Active cooling systems have a number of possible modes of failure including failures of pumps, valves, heat exchangers, distribution line or panel coolant passage restriction or rupture. The primary concern in detection of failures was associated with individual actively cooled structural panel failures such as cracks propagating into, or through, coolant tubes impact damage or indentation resulting in coolant flow restriction, or separation of a coolant tube from the actively cooled skin. The major problem in detection of a high out of tolerance skin temperature is the large surface area,  $2980 \text{ m}^2$  ( $32,134 \text{ ft}^2$ ) for the configuration studied.

The low heat load descent trajectory studies were initially based on abort from a nominal cruise dynamic pressure of  $24.1 \text{ kPa}$  ( $500 \text{ lbf/ft}^2$ ). This start of abort condition was retained for Mach 6, but was modified for Mach 4.5 and Mach 3 aborts to provide more realistic flight conditions. Aircraft angle of attack was limited to 20 degrees. Constraints were also placed on aircraft g-loads and passenger/personnel accelerations during abort.

A large number of potential candidate thermostructural concepts were screened and evaluated for possible application to the aircraft for within the Mach 3 to 6 flight regime. The structural temperatures were limited to a maximum of  $394 \text{ K}$  ( $250^\circ\text{F}$ ) during normal cruise and a maximum of  $478 \text{ K}$  ( $400^\circ\text{F}$ ) during abort. Material/structural life and maintenance requirements were important and, if possible without incurring major weight penalties, the structural temperature was held to under  $450 \text{ K}$  ( $350^\circ\text{F}$ ).

The "Fail-Safe Abort System" constituents for Mach 3, 4.5, and 6 cruise were integrated as systems, evaluated and conclusions were drawn.



### 3. BASELINE AIRCRAFT CHARACTERISTICS

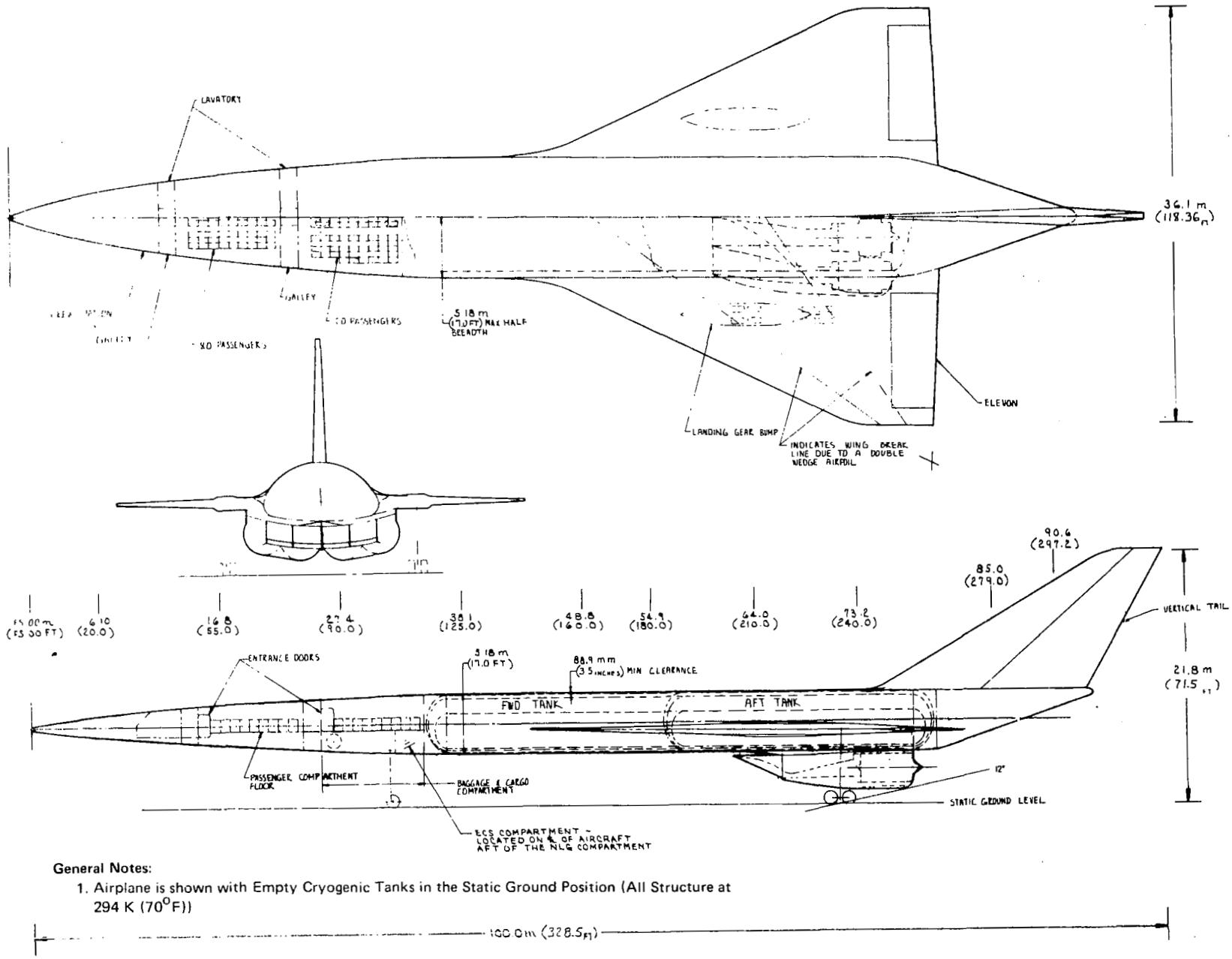
The baseline aircraft configuration used throughout the study was an actively cooled Mach 6 cruise design, identified in Reference (2) as Concept 3. The aircraft is a blended wing-body configuration, having an elliptical fuselage cross section, incorporating an integral "bubble" fuel tank structure. The aircraft was designed to carry 200 passengers over a range of 9.26 Mm (5000 NM). Takeoff fuel load was 108.9 Mg (240,000 lb<sub>m</sub>) of hydrogen. Figure 5 shows the general arrangement of the aircraft and indicates the pertinent wetted areas. The engine nacelle area was uncooled.

Complete details of the active cooling system design of the baseline aircraft are available in Reference (3). Figure 6 presents a thermodynamic summary for Concept 3. This system used methanol/water solution (60 percent methanol by weight) as the coolant. The coolant inlet temperature to the panels was 256 K (0°F). The coolant after absorbing the total heat load, was returned to the heat exchanger at approximately 294 K (70°F). This system was used as a reference active cooling system design and consisted of a nonredundant uninsulated (bare aluminum skin) system designed for a maximum aluminum structural temperature of 394 K (250°F).

The structural design details of the baseline aircraft are available in Reference (5). Aluminum honeycomb panels were used. Figure 7 shows, in some detail, the construction of the actively cooled panels used as moldline covering on the baseline aircraft. The honeycomb construction panel was selected because it resulted in a lighter aircraft.

Aerodynamic characteristics of the baseline aircraft were used for all three Mach number cruise conditions. Performance analyses of the baseline aircraft are available in Reference (2). Aerodynamic coefficients used to compute performance and aerodynamic characteristics, as well as the methods used to obtain them, are described therein. The propulsion system consists of four General Electric advanced hydrogen fueled turboramjet engines. The engine performance data is classified, Reference (6), and therefore is not included in this report. The baseline aircraft Mach number - altitude profile is presented in Figure 8.

## **FIGURE 5** **BASELINE AIRCRAFT GENERAL ARRANGEMENT**



Forward Fuselage	FS 0.00--34m	(FS 0.00--113.5 ft)	0.62 km <sup>3</sup> (22,000 ft <sup>3</sup> )
Center Fuselage	FS 34.6--80.0m	(FS 113.5--262.52 ft)	1.64 km <sup>3</sup> (65,100 ft <sup>3</sup> )
Aft Fuselage	FS 80.0--93.9m	(FS 262.52--308 ft)	0.11 km <sup>3</sup> (3,800 ft <sup>3</sup> )
Total Fuselage			2.57 km <sup>3</sup> (90,900 ft <sup>3</sup> )
Tank Volume			1.62 km <sup>3</sup> (57,200 ft <sup>3</sup> )
Tank Volume/Center Fuselage Volume			87.9%
Tank Volume/Total Fuselage Volume			63%

### Fuel Distribution

Tank Section	Type	Usable Volume*	Fuel Weight
Forward Fuselage	Integral	0.76 km <sup>3</sup> (26,778 ft <sup>3</sup> )	53.7 Mg (118,400 lb)
Aft Fuselage	Integral	0.78 km <sup>3</sup> (27,522 ft <sup>3</sup> )	55.2 Mg (121,600 lb)
Total		1.54 km <sup>3</sup> (54,300 ft <sup>3</sup> )	108.9 Mg (240,000 lb)

\*5% of tank volume allowed for ullage, rings, etc. Usable volume = 0.95 tank volume

Fuel Liquid hydrogen @ 20.3 K (-423°F)  $\rho$  (density) = 70.8 Kg/m<sup>3</sup> (4.42 lb/ft<sup>3</sup>)

### Physical Characteristics

Item	Wing	Vertical Tail
S <sub>theo</sub>	0.96 km <sup>2</sup> (10,377 ft <sup>2</sup> )	0.14 km <sup>2</sup> (1,535 ft <sup>2</sup> )
AR	1.35	—
$\lambda$	0.15	—
b	36.1 m (118.36 ft)	11.9 m (39.18 ft)
b/2	18.0 m (59.18 ft)	—
CR	46.5 m (152.47 ft)	18.6 m (61.01 ft)
C <sub>T</sub>	7.0 m (22.87 ft)	5.1 m (16.66 ft)
MAC	31.6 m (103.64 ft)	13.3 m (43.49 ft)
$\Delta$ LE (deg)	65	60
$\Delta$ TE (deg)	-3	30
Incidence (deg)	+1/2	—
Dihedral	0	—
Thickness Ratio	0.03	0.03

### Performance Summary

Range	9.20 Mg (4,968 NM)
Payload (200 Passengers)	21.8 Mg (48,000 lb)
Operating Weight Empty	187.3 Mg (412,816 lb)
Takeoff Gross Weight	296.1 Mg (652,816 lb)

### Propulsion

(4) GE5/JZ6-C 400 kN (90,000 lb) T-SLS per Engine Uninstalled
Total Inlet Capture Area (A <sub>Ctotal</sub> ) = 15.8 m <sup>2</sup> (170 ft <sup>2</sup> )

	Tire Size
Main Gear	1.27 m x 0.51 m (50 in. x 20 in.)
Nose Gear	1.27 m x 0.51 m (50 in. x 20 in.)

### Wetted Area

Fuselage	1.63 km <sup>2</sup> (17,600 ft <sup>2</sup> )
Nacelle	0.38 km <sup>2</sup> (4,080 ft <sup>2</sup> )
Wing	1.07 km <sup>2</sup> (11,464 ft <sup>2</sup> )
Vertical Tail	0.28 km <sup>2</sup> (3,070 ft <sup>2</sup> )
Total <sup>1</sup>	3.36 km <sup>2</sup> (36,214 ft <sup>2</sup> )

Fineness Ratio	13.10
Total Aircraft Volume	3.5 km <sup>3</sup> (123,800 ft <sup>3</sup> )
Planform Area	1.28 km <sup>2</sup> (13,756 ft <sup>2</sup> )
Max Cross Section Area	98.5 m <sup>2</sup> (1,060 ft <sup>2</sup> )
Less Capture Area	15.8 m <sup>2</sup> (170 ft <sup>2</sup> )
Net Cross Sectional Area	82.7 m <sup>2</sup> (890 ft <sup>2</sup> )
Mach No. (Cruise)	6
V <sup>2/3</sup> / S <sub>p</sub> Factor	0.156

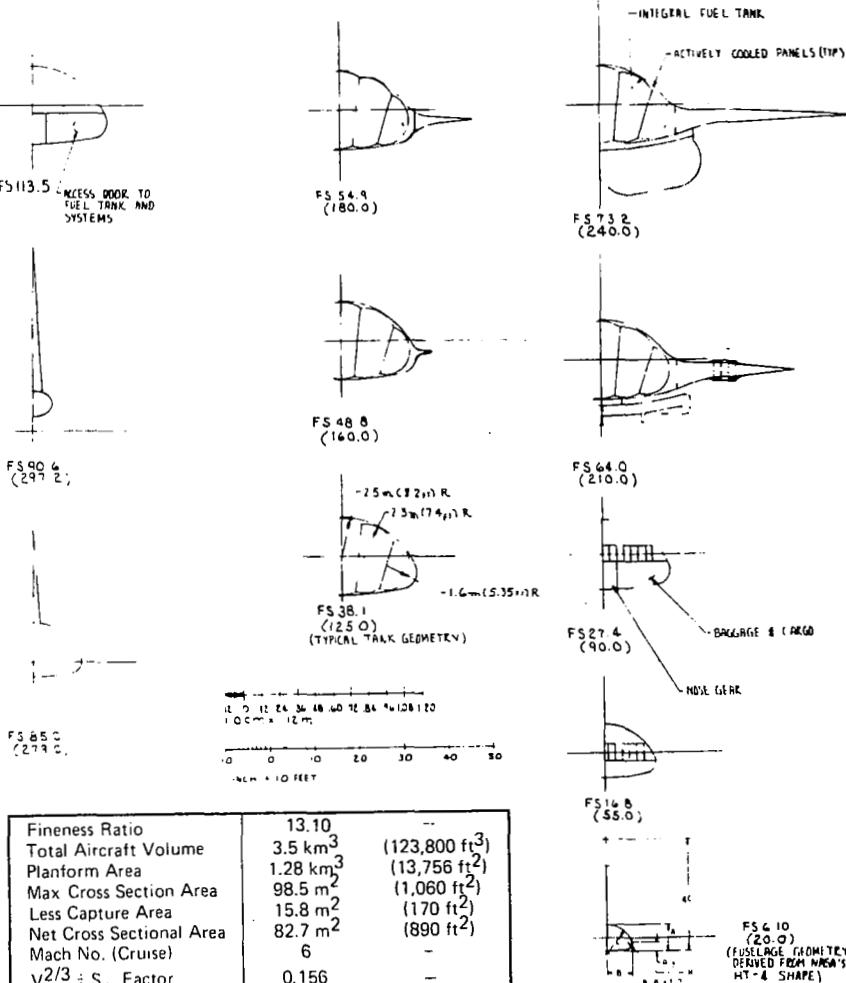
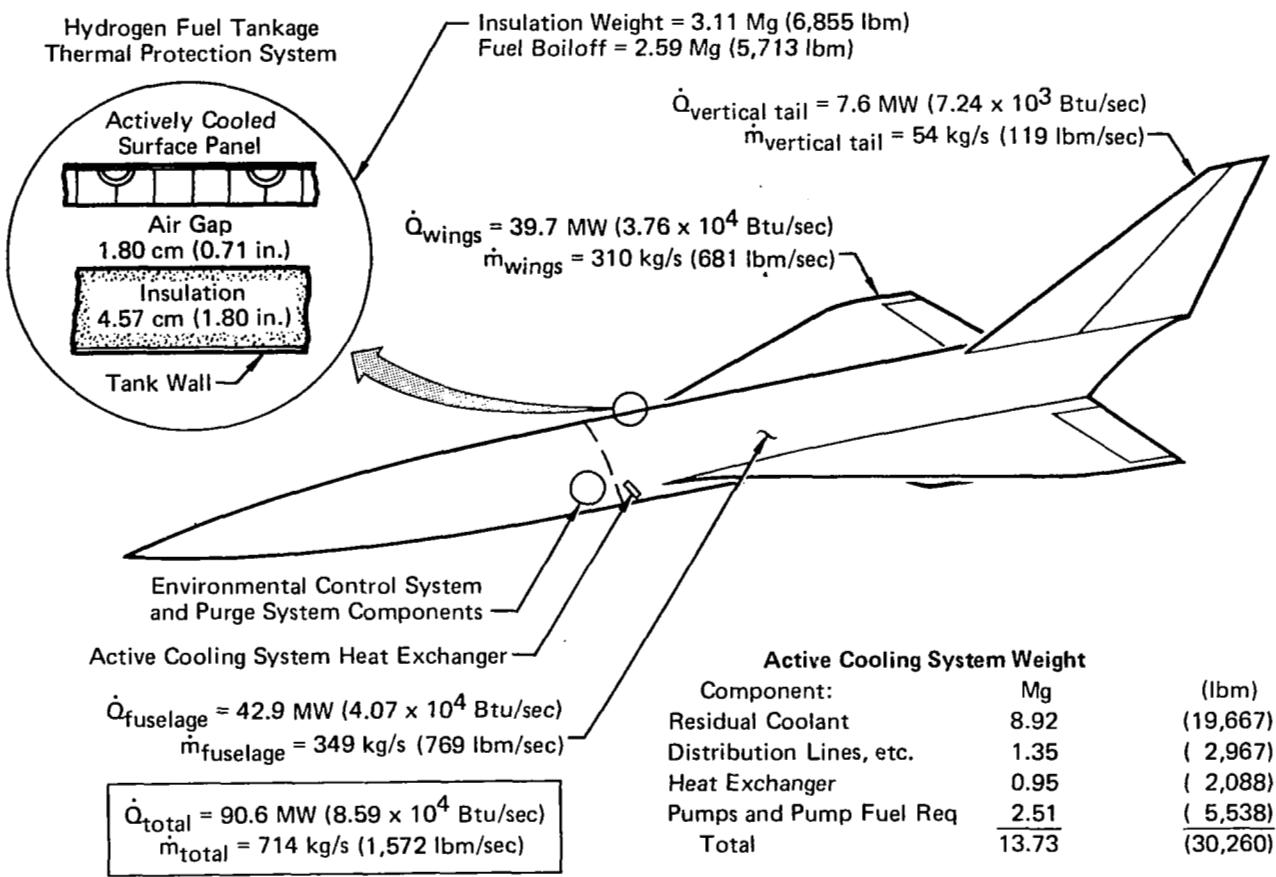
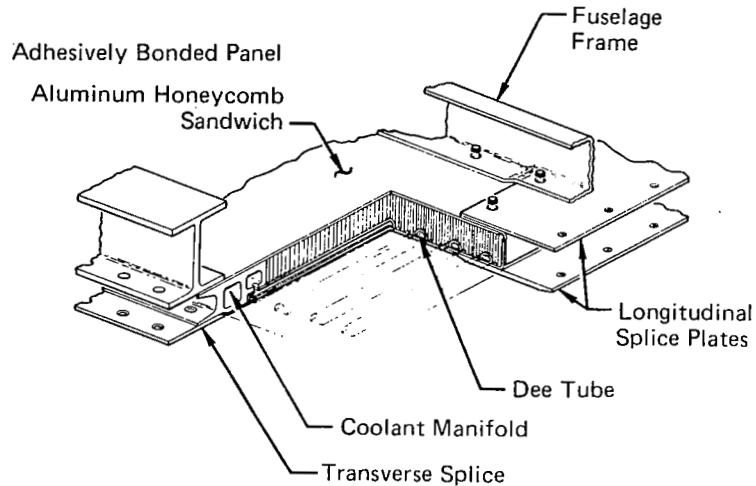


FIGURE 5 (Continued)  
BASELINE AIRCRAFT GENERAL ARRANGEMENT

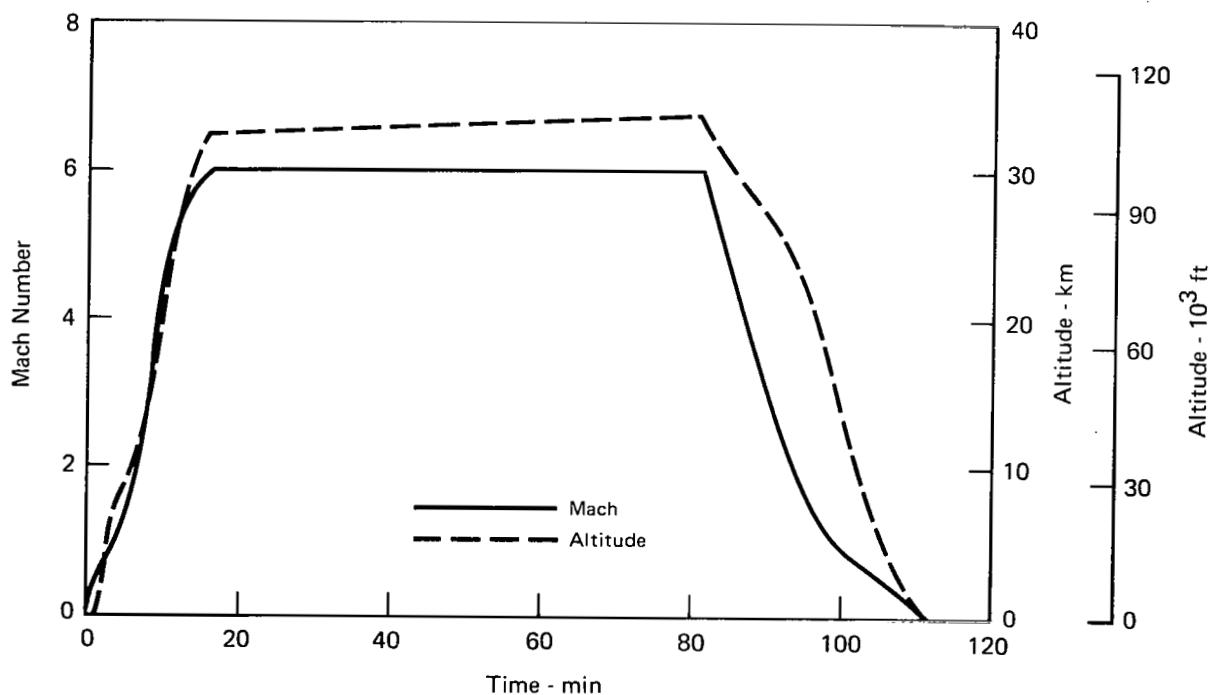




**FIGURE 6**  
**THERMODYNAMIC SUMMARY MACH 6 BASELINE**



**FIGURE 7**  
ACTIVELY COOLED PANEL CONSTRUCTION



**FIGURE 8**  
MACH NUMBER AND ALTITUDE vs TIME, BASELINE AIRCRAFT

#### 4. PRESENTATION OF RESULTS

The study to conceptually design and evaluate a fail-safe actively cooled structural system, capable of detecting failures, and utilizing passive thermostructural concepts to absorb the abort descent heat load involved four interacting tasks.

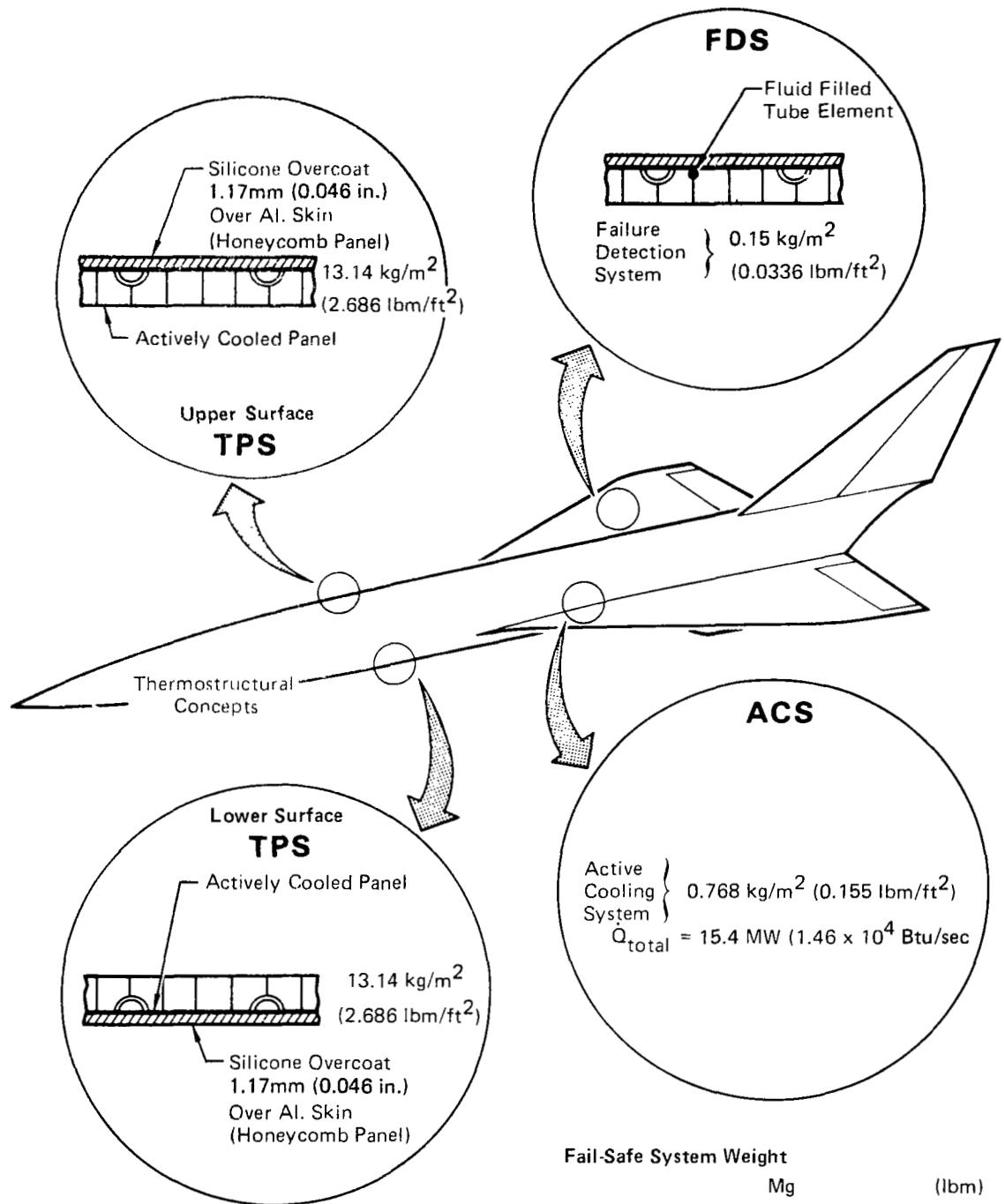
These tasks were, in broad terms; (1) failure detection, (2) abort trajectory optimization, (3) fail-safe abort thermostructural concepts, and (4) abort system integration and optimization. The prime results of the study were the output of task (4), and thus, are presented first. Then the results of task (1) (failure detection) are presented. These are followed by task (2) (abort trajectories), and task (3) (thermostructural concepts) results.

##### 4.1 SELECTED FAIL-SAFE SYSTEMS

The selected fail-safe system concepts for each of the three typical cruise Mach number conditions are presented. The selected concepts include the three basic factors required to assure safety of flight: A failure detection system capable of detecting failures or malfunctions in the active cooling system as well as failures in individual actively cooled panels; an abort trajectory capable of providing a near minimum heat load to the aircraft during descent from cruise Mach number; and thermostructural design approaches which minimize both the weight and the maximum structural temperature during an abort descent.

The systems for each cruise Mach number are discussed in following paragraphs.

- o Mach 3 Fail-Safe System - Figure 9 presents a summary of the Mach 3 system. Ancillary component unit weights are given as well as total system weight. As shown, both the upper and lower surfaces of the aircraft are overcoated with a thin silicone elastomer. This provides both a reduction in cruise heat transfer to the actively cooled structural panels and heat protection during a cooling system failure induced abort.



#### Fail-Safe System Weight

	Mg	(lbm)
Actively Cooled Structure	37.7	(83,227)
(TPS) Thermal Protection Systems	1.5	( 3,093)
(ACS) Active Cooling System	2.3	( 4,974)
(FDS) Failure Detection System	0.5	( 1,081)
Total	42.0	(92,375)

**FIGURE 9**  
**MACH 3 FAIL-SAFE SYSTEM SUMMARY**

The fail-safe system selected for Mach 3 cruise utilized conventional instrumentation for monitoring of active cooling system parameters to provide warning of system malfunction or failure. The failure detection system components used to detect failures (loss of cooling) of individual actively cooled panels were Freon filled tube elements.

These elements (sensors) have the potential to provide a continuous record of average panel temperature. With loss of panel cooling (or individual coolant tube) the Freon vaporizes, providing a pressure pulse signal to a pressure transducer. The system of tube elements use a common Freon reservoir to provide volume for thermal expansion of the Freon. Pressure is maintained at a constant level by a compensator within the liquid Freon reservoir during normal operation. The pressure pulse is experienced only with boiling of the Freon due to loss of cooling. Sensor response times were estimated at less than 15 seconds for both the upper and lower surfaces of the aircraft. These response times were judged more than adequate for the structural heat protection provided.

The abort descent trajectory used provided a low heat load to the aircraft during abort. This trajectory used a constant g-load pull-up to a high-lift coefficient glide condition. Engine power was cut at initiation of abort.

The thermostructural concept selected, as shown by Figure 9, was a silicone insulative overcoat over aluminum honeycomb sandwich containing coolant manifolds and dee-shaped tubes. The coating thickness was an average of 1.17 mm (0.046 in). Maximum temperatures of the aluminum structure during abort were well under the reuse limit of 450 K (350°F). Temperatures of the silicone during cruise were 414 K (285°F) for the upper surface and 451 K (353°F) on the lower surface.

Figure 10 presents a comparison of the fail-safe system with a Mach 3 bare aluminum skin baseline system. As shown, the fail-safe capability resulted in a fixed weight increase of 0.5 Mg (847 lbm) over the total weight of the baseline system, about  $168 \text{ g/m}^2$  ( $0.0264 \text{ lbm/ft}^2$ ). However, when the hydrogen heat sink requirements are considered, the fail-safe system has the lowest total weight. The baseline system would need 32.1 Mg (70,900 lbm) of excess hydrogen for cooling during the 146 minutes required to cruise 7.41 Mm (4000 NM).

System Weights - Mg (lbm) ▲1		
	Baseline System	Fail-Safe System
Structure ▲2	38.2 (84,243)	39.2 (86,320)
Active Cooling System ▲3	3.3 (7,285)	2.3 (4,974)
Failure Detection System ▲4	None	0.5 (1,081)
Subtotal	41.5 (91,528)	42.0 (92,375)
Additional Hydrogen Required for Cooling ▲5	32.1 (70,900)	None
Total	73.6 (162,428)	42.0 (92,375)
	Heat Loads - MW (Btu/sec) ▲1	
Cooling System Heat Load (Start of Cruise) ▲6	25 ( $2.37 \times 10^4$ )	15.4 ( $1.46 \times 10^4$ )
Hydrogen Fuel Flow Heat Sink Capacity (Cruise) ▲7	15.4 ( $1.46 \times 10^4$ )	15.4 ( $1.46 \times 10^4$ )

Notes: ▲1 All weights (and heat loads) based on equal aircraft configurations with  $2980\text{m}^2$  ( $32,134\text{ ft}^2$ ) of cooled surface area. Values shown are derived from average values for  $1618\text{m}^2$  ( $17,449\text{ ft}^2$ ) with cruise heating rate equal to typical upper surface average and  $1362\text{m}^2$  ( $14,685\text{ ft}^2$ ) with cruise heating rate equal to typical lower surface average.

▲2 Actively cooled panels, attachments, non-optimums, heat shields and insulation (or other heat protection) where applicable.

▲3 All components, instrumentation, coolant, coolant distribution lines, etc.

▲4 All components integrated into aircraft.

▲5 Based on ▲6 ▲7 for Mach 3 Cruise for 7.41 Mm (4000 NM). Does not include hydrogen containment.

▲6 Baseline has unprotected aluminum skin at average of  $366\text{K}$  ( $200^\circ\text{F}$ ) at cruise.

▲7 Based on typical values for duct burning turbofan engines,  $\Delta T_f = 33\text{K}$  to  $311\text{K}$  ( $500^\circ\text{F}$ )

**FIGURE 10  
COMPARISON OF MACH 3 CRUISE SYSTEMS**

Thus, the differences between the baseline and the fail-safe system, in terms of system heat load versus available hydrogen fuel heat sink capacity, are significant. This translates into a weight difference of about 31.7 Mg (70,000 lbm) in favor of the silicone overcoated fail-safe system. Weight for containment of the excess hydrogen in the baseline aircraft is not included in this weight comparison.

The baseline aircraft, with the bare unprotected aluminum skin, would not be capable of operation unless a large portion of the cooled area were heat shielded in some manner, with an attendant increase in weight. The fail-safe system heat load matches the heat sink availability and is a viable system for Mach 3 operation.

o Mach 4.5 Fail-Safe System - The selected Mach 4.5 cruise fail-safe system was configured with the same failure detection devices as the Mach 3 system. The same type of abort descent trajectory was utilized to reduce descent heat load.

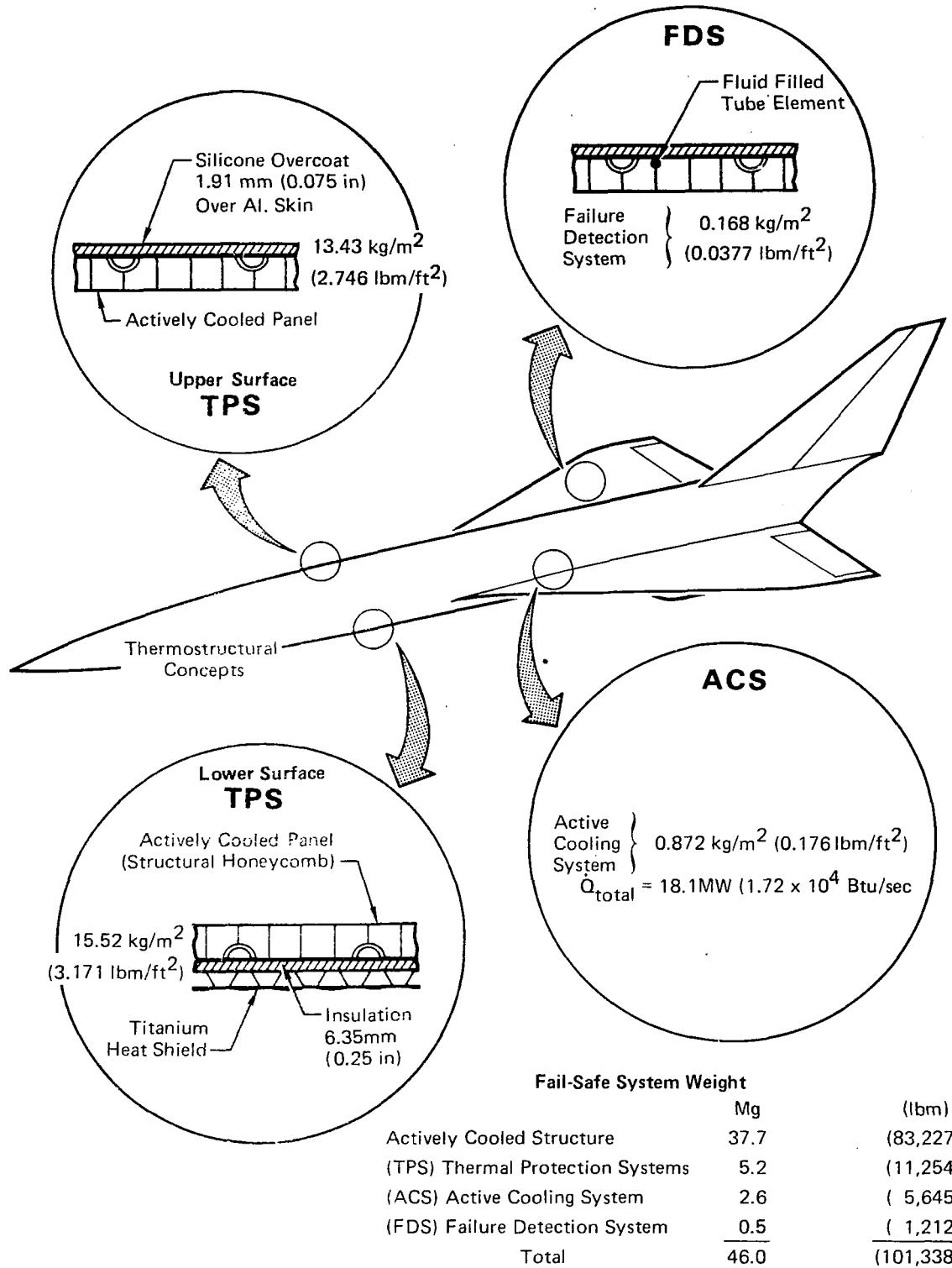
The thermostructural approach on this aircraft was to use the silicone overcoat material on the upper surfaces (average thickness of 1.91 mm (0.075 in)) and a titanium heat shield, with insulation (6.35 mm (0.25 in) thick), on the lower surfaces. Maximum abort temperatures were under the limit of 450 K (350°F) for reuse of the aluminum structure.

Figure 11 presents a summary of the Mach 4.5 system weight. As shown, the passive heat protection represents about 11 percent of the total. The silicone elastomer overcoated upper surface and the corrugation stiffened beaded skin titanium heat shield-insulation package on the lower surface are highly efficient in reducing the amount of heat absorbed by the hydrogen fuel. The lower surface TPS is assembled with the insulation packages against the aluminum skins to avoid potential boundary layer leakage flow paths.

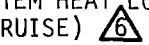
Figure 12 compares the fail-safe system with the bare unprotected aluminum skin baseline system. In this case, the addition of fail-safe capability increased the fixed weight by 1.7 Mg (3734 lbm), or  $0.57 \text{ kg/m}^2$  ( $0.116 \text{ lbm/ft}^2$ ), compared to the no-abort-capability baseline.

The baseline system heat load, at start of cruise at Mach 4.5, was approximately 123% of available fuel heat sink capacity. The fail-safe system heat load was about 41% of fuel heat sink capacity. As shown in Figure 12, the baseline aircraft would require an additional 10.7 Mg (23,600 lbm) of excess hydrogen for cooling during the 98 minutes required to cruise 7.41 Mm (4000 NM). In this case, the fail-safe design would reduce the total weight nearly 9.1 Mg (20,000 lbm).

During cruise, the fail-safe system TPS temperature levels are moderate. The silicone overcoat (upper surface) has a steady state level of 533 K (499°F). The lower surface heat shields experience 663 K (734°F).



**FIGURE 11**  
**MACH 4.5 FAIL-SAFE SYSTEM SUMMARY**

	SYSTEM WEIGHTS - Mg (lbm) 	
	BASELINE SYSTEM	FAIL-SAFE SYSTEM
STRUCTURE 	38.7 (85,303)	42.9 (94,481)
ACTIVE COOLING SYSTEM 	5.6 (12,307)	2.6 (5,645)
FAILURE DETECTION SYSTEM 	NONE	0.5 (1,212)
SUBTOTAL	44.3 (97,604)	46.0 (101,338)
ADDITIONAL HYDROGEN REQUIRED FOR COOLING 	10.7 (23,600)	NONE
TOTAL	55.0 (121,204)	46.0 (101,338)
	HEAT LOADS - MW (Btu/sec) 	
COOLING SYSTEM HEAT LOAD (START OF CRUISE) 	54.7 ( $5.19 \times 10^4$ )	18.1 ( $1.72 \times 10^4$ )
HYDROGEN FUEL FLOW HEAT SINK CAPACITY (CRUISE) 	44.4 ( $4.21 \times 10^4$ )	44.4 ( $4.21 \times 10^4$ )

NOTES:  ALL WEIGHTS AND HEAT LOADS BASED ON EQUAL AIRCRAFT CONFIGURATIONS WITH  $2980 \text{ m}^2$  ( $32,134 \text{ ft}^2$ ) OF COOLED SURFACE AREA. VALUES SHOWN ARE DERIVED FROM AVERAGE VALUES FOR  $1618 \text{ m}^2$  ( $17,449 \text{ ft}^2$ ) WITH CRUISE HEATING RATE EQUAL TO TYPICAL UPPER SURFACE AVERAGE AND  $1362 \text{ m}^2$  ( $14,685 \text{ ft}^2$ ) WITH CRUISE HEATING RATE EQUAL TO TYPICAL LOWER SURFACE AVERAGE.

-  ACTIVELY COOLED PANELS, ATTACHMENTS, NON-OPTIMUMS, HEAT SHIELDS AND INSULATION (OR OTHER HEAT PROTECTION) WHERE APPLICABLE.
-  ALL COMPONENTS, INSTRUMENTATION, COOLANT, COOLANT DISTRIBUTION LINES, ETC.
-  ALL COMPONENTS INTEGRATED INTO AIRCRAFT
-  BASED ON   FOR MACH 4.5 CRUISE FOR 7.41 Mm (4000 NM). DOES NOT INCLUDE HYDROGEN CONTAINMENT.
-  BASELINE HAS UNPROTECTED ALUMINUM SKIN AT AVERAGE OF 366 K ( $200^\circ\text{F}$ ) AT CRUISE.
-  BASED ON TYPICAL VALUES FOR TURBORAMJET ENGINES (RAMJET MODE AT CRUISE)  $\Delta T_f = 33 \text{ K}$  TO  $311 \text{ K}$  ( $500^\circ\text{F}$ ).

**FIGURE 12  
COMPARISON OF MACH 4.5 CRUISE SYSTEMS**

Maximum temperatures during abort were low, about 409 K ( $276^\circ\text{F}$ ) for the upper surface aluminum structure and 537 K ( $507^\circ\text{F}$ ) on the silicone. The lower surface structure reached 420 K ( $296^\circ\text{F}$ ) and the heat shield 677 K ( $759^\circ\text{F}$ ). With these maximum temperatures, all thermostructural components were judged reusable.

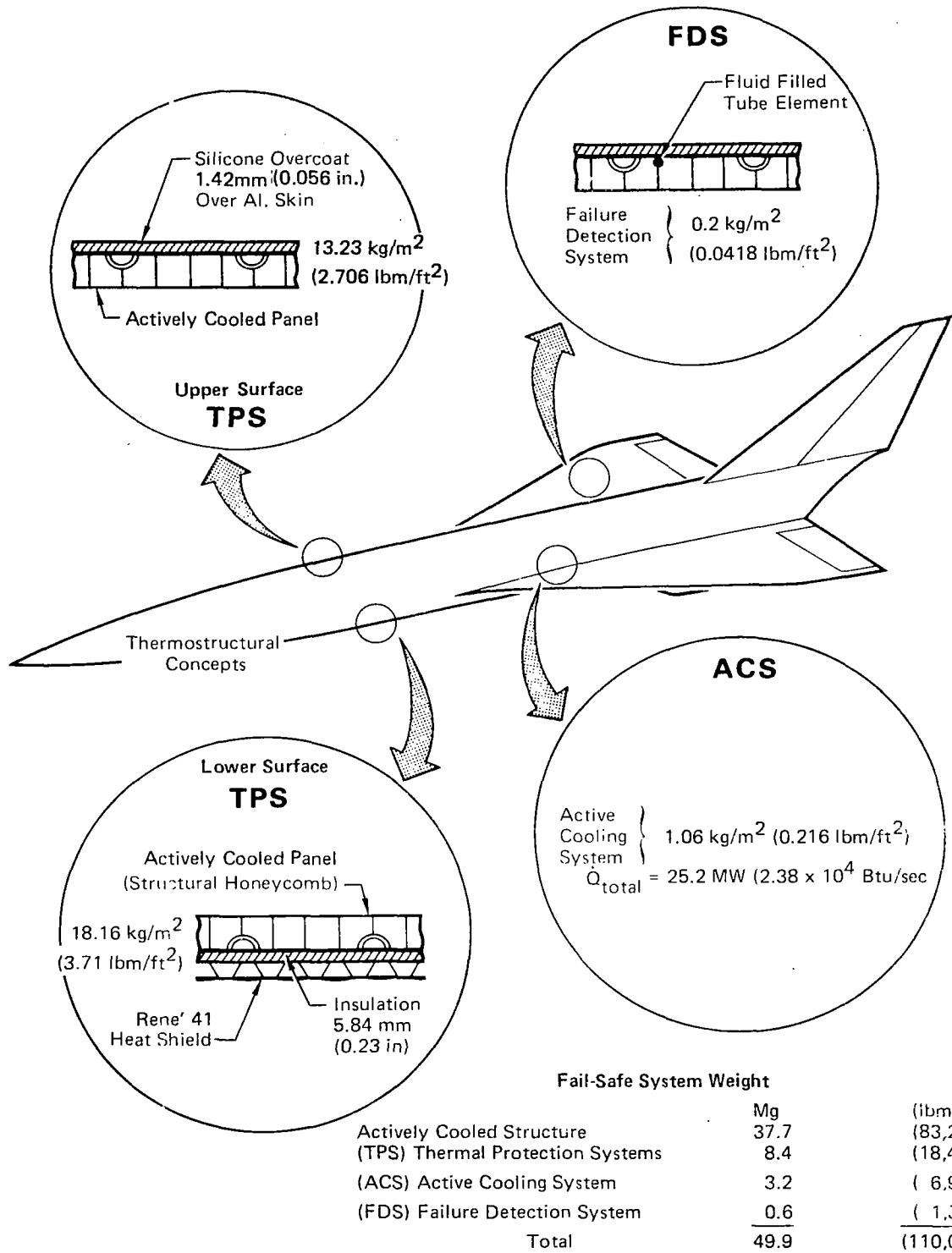
o Mach 6 Fail-Safe System - This system again used the same type of failure detection methods and abort trajectory as the Mach 3 and 4.5 systems. The system weights are presented in Figure 13.

A silicone overcoated (1.42 mm (0.056 in) thickness) upper surface and a Rene' 41 heat shielded (and insulated) lower surface were combined to provide abort heat protection of the aluminum actively cooled panels, and to provide a reduction in absorbed heat flux at cruise conditions. The maximum structural temperatures during abort (total loss of cooling) would be approximately 443 K (337°F) for the upper surfaces and 465 K (377°F) for the lower surface.

The lower surface structural temperatures could be reduced to the limit for reuse (450 K (350°F)) with some additional lower surface insulation. The weight penalty for this additional insulation would be low. The temperatures of the TPS components are well within the capabilities of the materials. At cruise the silicone is at 554 K (537°F) and reaches 563 K (554°F) during the abort. The lower surface heat shields are, typically, at temperatures of 850 K (1070°F) at cruise conditions and 866 K (1100°F) maximum during abort.

Figure 14 illustrates the weight differences between the fail-safe system and the unprotected baseline system. Fail-safe capability resulted in a fixed weight increase, compared to the baseline weight, of 3.3 Mg (7162 lbm), about 1.11 kg/m<sup>2</sup> (0.229 lbm/ft<sup>2</sup>). System heat loads for the unprotected baseline were 167% of available fuel heat sink capacity at start of cruise conditions. Consideration of excess hydrogen required for cooling increases the baseline aircraft weight by at least 29.4 Mg (64,700 lbm). In this case, the fail-safe design total weight is about 26.1 Mg (57,500 lbm) less than the baseline. The fail-safe system requires only 46.5% of available fuel heat sink capacity.

Figure 15 presents a comparison of Reference (7) results and this study. As shown, the "Fail-Safe Abort System" total unit weight is approximately 90% of the unit weight of the Reference (7) unshielded 366 K (200°F) structure with redundant cooling systems, which requires 10.44 Mg (23,000 lbm) of excess hydrogen for cooling for a cruise range of approximately 5.87 Mm (3170 NM). The additional 4.54 Mg (10,000 lbm) required for containment of the excess hydrogen is not included in the weight summary.



**FIGURE 13**  
**MACH 6 FAIL-SAFE SYSTEM SUMMARY**

	SYSTEM WEIGHTS - Mg (lbm) 	
	BASELINE SYSTEM	FAIL-SAFE SYSTEM
STRUCTURE 	38.9 (85,734)	46.1 (101,703)
ACTIVE COOLING SYSTEM 	7.7 (17,105)	3.2 (6,955)
FAILURE DETECTION SYSTEM 	NONE	0.6 (1,343)
SUBTOTAL	46.6 (102,839)	49.9 (110,001)
ADDITIONAL HYDROGEN REQUIRED FOR COOLING 	29.4 (64,700)	NONE
TOTAL	76.0 (167,539)	49.9 (110,001)
	HEAT LOADS- MW (Btu/sec) 	
COOLING SYSTEM HEAT LOAD (START OF CRUISE) 	90.6 ( $8.59 \times 10^4$ )	25.2 ( $2.39 \times 10^4$ )
HYDROGEN FUEL FLOW HEAT SINK CAPACITY (CRUISE) 	54.2 ( $5.14 \times 10^4$ )	54.2 ( $5.14 \times 10^4$ )

- NOTES:
-  ALL WEIGHTS (AND HEAT LOADS) BASED ON EQUAL AIRCRAFT CONFIGURATIONS WITH  $2980 \text{ m}^2$  ( $32,134 \text{ ft}^2$ ) OF COOLED SURFACE AREA. VALUES SHOWN ARE DERIVED FROM AVERAGE VALUES FOR  $1618 \text{ m}^2$  ( $17,449 \text{ ft}^2$ ) WITH CRUISE HEATING RATE EQUAL TO TYPICAL UPPER SURFACE AVERAGE AND  $1362 \text{ m}^2$  ( $14,685 \text{ ft}^2$ ) WITH CRUISE HEATING RATE EQUAL TO TYPICAL LOWER SURFACE AVERAGE.
  -  ACTIVELY COOLED PANELS, ATTACHMENTS, NON-OPTIMUMS, HEAT SHIELDS AND INSULATION (OR OTHER HEAT PROTECTION) WHERE APPLICABLE.
  -  ALL COMPONENTS, INSTRUMENTATION, COOLANT, COOLANT DISTRIBUTION LINES, ETC.
  -  ALL COMPONENTS INTEGRATED INTO AIRCRAFT
  -  BASED ON   FOR MACH 6 CRUISE FOR 7.41 Mm (4000 NM). DOES NOT INCLUDE HYDROGEN CONTAINMENT.
  -  BASELINE HAS UNPROTECTED ALUMINUM SKIN AT AVERAGE OF 366 K ( $200^\circ\text{F}$ ) AT CRUISE.
  -  BASED ON TYPICAL VALUES FOR TURBORAMJET ENGINES (RAMJET MODE AT CRUISE)  $\Delta T_f = 33 \text{ K}$  TO  $311 \text{ K}$  ( $500^\circ\text{F}$ ).

FIGURE 14  
COMPARISON OF MACH 6 CRUISE SYSTEMS

Unit Weight - kg/m <sup>2</sup> (lbm/ft <sup>2</sup> )			
	Reference ⑦ Unshielded 366K (200°F) Structure, Redundant Cooling System	Reference ⑦ Shielded (33%) 366K (200°F) Structure, Redundant Cooling System	Fail Safe System 366K (200°F) Structure (ALL SHIELDED)
Structure	① 12.651 (2.5912)	① 12.651 (2.5912)	② 12.650 (2.5900)
Cooling System	2.697 (0.5524)	1.798 (0.3682)	1.057 (0.2164)
Failure Detection System	None	None	0.204 (0.0418)
Other	① 3.303 (0.6765) ③	① 1.594 (0.3265) ④	⑤ 2.802 (0.5749)
Subtotal	18.651 (3.8201)	16.043 (3.2859)	16.713 (3.4231)
Additional Hydrogen Heat Sink ⑥	0.860 (0.1760)	0.039 (0.0080)	None
Total	19.511 (3.9961)	16.082 (3.2939)	16.713 (3.4231)

Notes: ① 3159 m<sup>2</sup> (34,000 ft<sup>2</sup>) of cooled surface, cruise range 5.87 Mm (3170 NM)

② 2980 m<sup>2</sup> (32,134 ft<sup>2</sup>) of cooled surface, range 7.41 Mm (4000 NM)

③ Excess hydrogen [10.44 Mg (23,000 lbm) for cooling, does not include hydrogen containment.

④ Heat shielding [4.5359 Mg (10,000 lbm)] plus 0.4539 Mg (1000 lbm) excess hydrogen for cooling, does not include hydrogen containment.

⑤ Includes Rene' 41 heat shields on lower surface, silicone elastomer overcoat on upper surfaces.

⑥ Additional hydrogen for cooling for Δ time (0.23 hours) to give total cruise range of 7.41 Mm (4,000 NM). Does not include hydrogen containment.

### FIGURE 15 MACH 6 CRUISE TRANSPORT AIRCRAFT WEIGHT SUMMARY

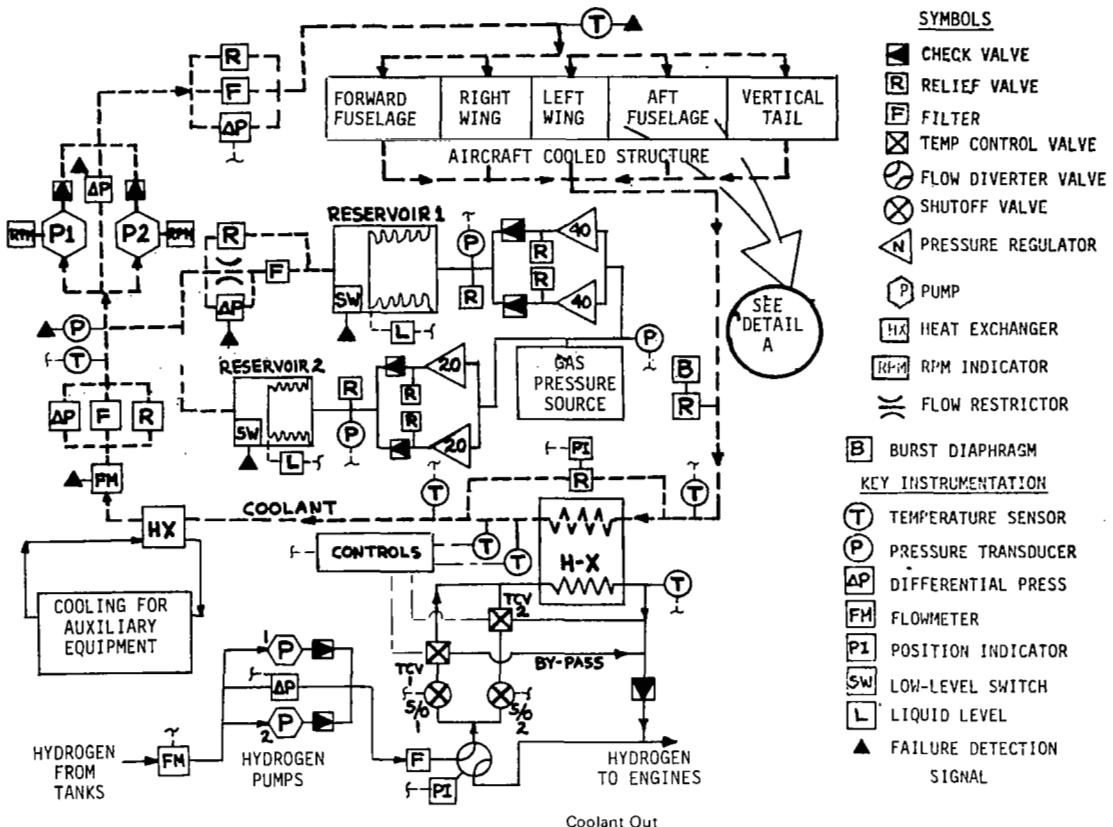
The fail-safe system aircraft has a cruise range of over 7.41 Mm (4000 nm). Therefore, to place these concepts on nearly equal ground, additional weight (hydrogen) was added to provide the required cooling for the 0.23 hours additional time required to reach 7.41 Mm (4000 nm). This excess hydrogen is shown in Figure 15 on a unit area basis.

The "Fail-Safe" concept does not require any additional hydrogen for cooling, using only 46.5% of available fuel heat sink at start of cruise conditions. The Reference (7) shielded structure (about 33% of area is shielded) concept with redundant cooling systems is shown in Figure 15 to have lower unit weight than the "Fail-Safe" system concept. However, this system does not have the abort capability, or the capability to detect loss of cooling within the individual actively cooled panels. Loss of cooling could result in structural failure if undetected. Addition of heat protection to the unshielded areas would add approximately  $0.619 \text{ kg/m}^2$  ( $0.1268 \text{ lbm/ft}^2$ ) to the presented weights of the Reference (7) shielded concept for a total unit weight of  $16.697 \text{ kg/m}^2$  ( $3.4207 \text{ lbm/ft}^2$ ). This weight increase considers the cooling system weight reductions due to reduced heat load and the elimination of excess hydrogen. A comparison of this unit weight with the "Fail-Safe" system unit weight of  $16.713 \text{ kg/m}^2$  ( $3.423 \text{ lbm/ft}^2$ ) indicates that "Fail-Safe" capability results in unit weights approximately equal to those of an equivalent all shielded concept with redundant cooling systems.

#### 4.2 FAILURE DETECTION

The study of the baseline active cooling system, Figure 16, did not result in any unique instrumentation requirements other than instrumentation required to measure, and provide a warning of, actively cooled structural panel over-temperature. Therefore, the primary effort was directed toward methods for detecting individual panel "hot spots".

Approximately twenty different candidate approaches to detection of the loss of cooling of individual panels, or areas on an individual panel, were devised and evaluated. The approach retained as the most promising was the Freon filled tube elements. Figure 17 illustrates the sensor assembly. A small reservoir, with pressure transducers and a pressure relief valve, maintains constant system pressure and provides volume for thermal expansion of the contained fluid. During normal operation the liquid level in the reservoir provides a continuous record of average panel temperature. At higher than normal temperatures, the contained liquid boils and provides a pressure signal. The sensor element circuit reservoir pressure relief valve prevents pressure from exceeding safe levels. An alternate approach, eutectic salt elements, which provides only a warning of overtemperature without continuous temperature monitoring capability is shown in Figure 18.



#### SYMBOLS

█	CHECK VALVE
█	RELIEF VALVE
█	FILTER
█	TEMP CONTROL VALVE
○	FLOW DIVERTER VALVE
✗	SHUTOFF VALVE
△	PRESSURE REGULATOR
○ P	PUMP
○ HX	HEAT EXCHANGER
○ RPM	RPM INDICATOR
—	FLOW RESTRICTOR
B	BURST DIAPHRAGM

#### KEY INSTRUMENTATION

○ T	TEMPERATURE SENSOR
○ P	PRESSURE TRANSDUCER
○ AP	DIFFERENTIAL PRESS
○ FM	FLOWMETER
○ PI	POSITION INDICATOR
○ SW	LOW-LEVEL SWITCH
○ L	LIQUID LEVEL
△	FAILURE DETECTION SIGNAL

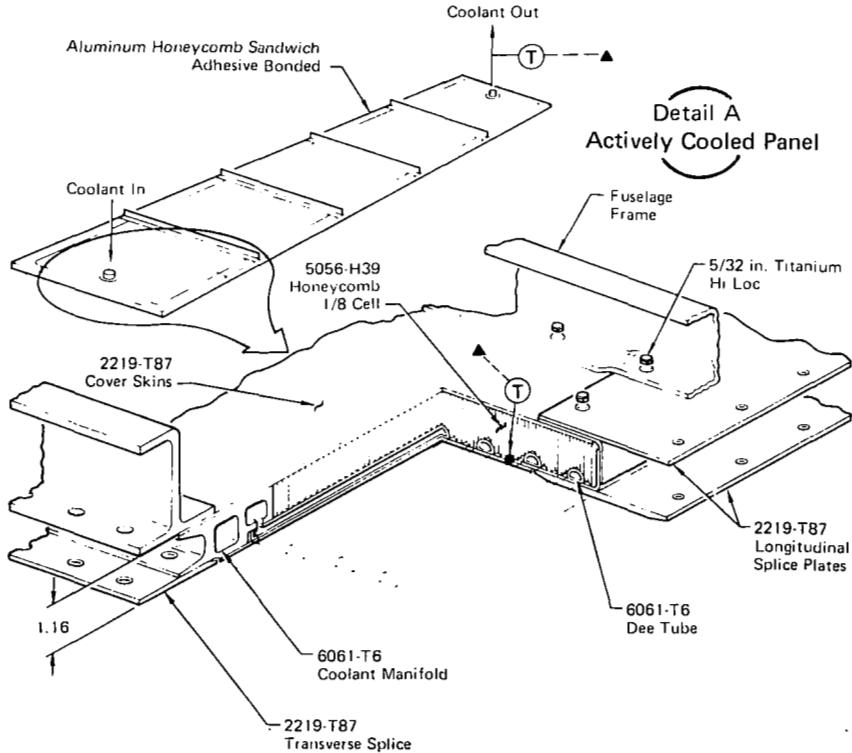
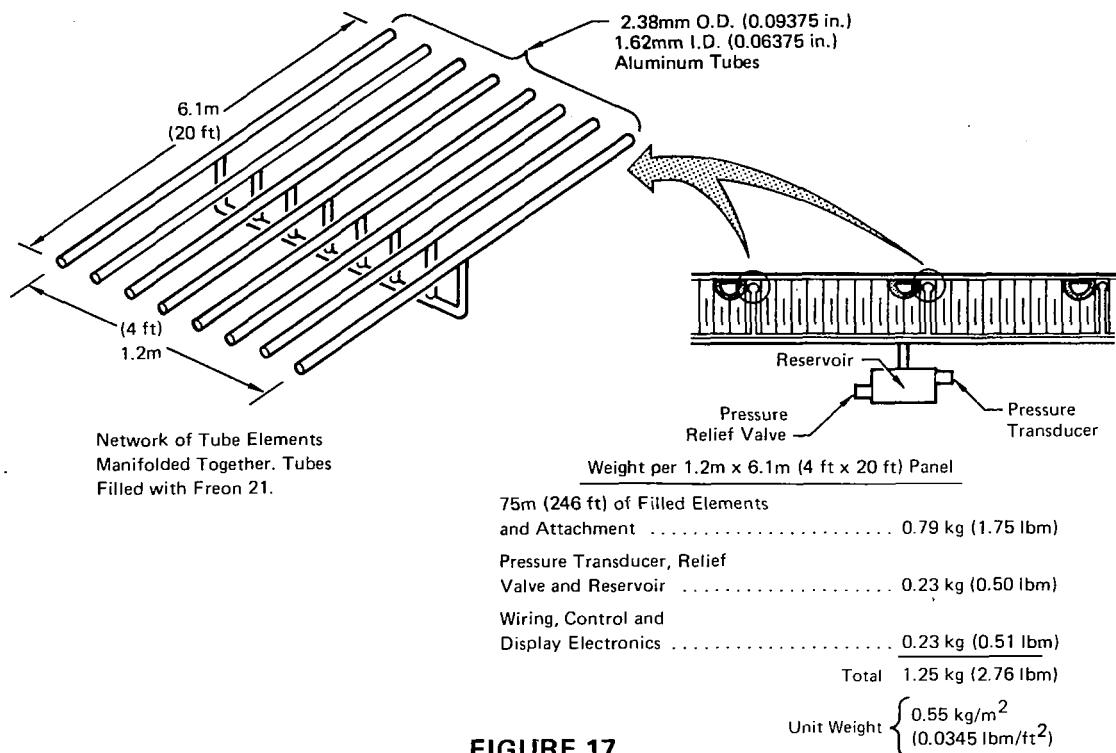
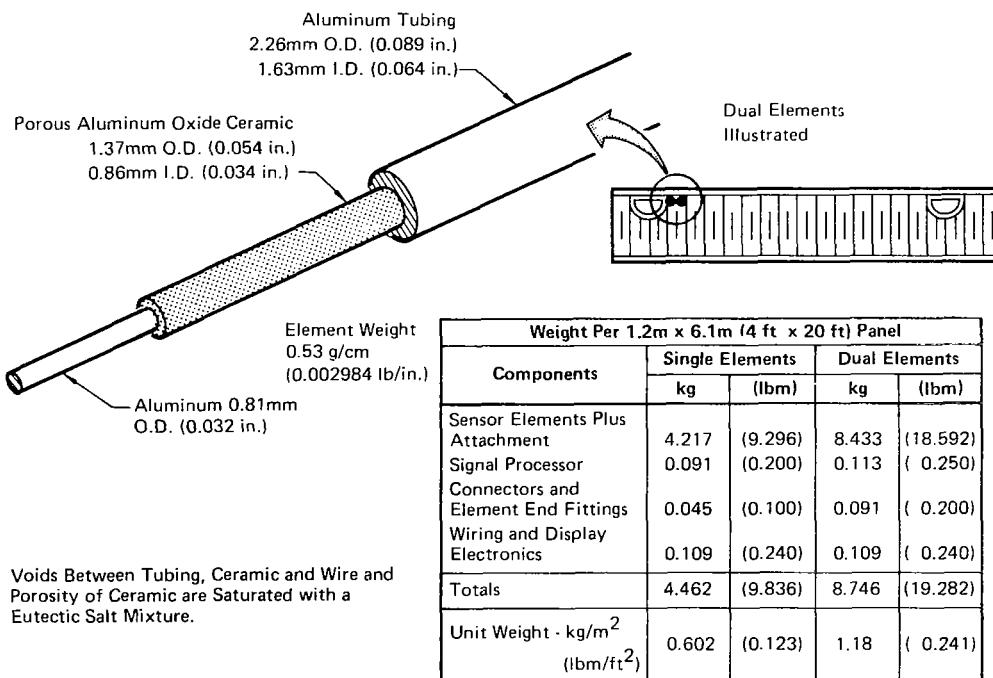


FIGURE 16  
BASELINE ACTIVE COOLING SYSTEM SCHEMATIC



**FIGURE 17**  
**FLUID FILLED TUBE ELEMENTS (PHASE CHANGE)**



**FIGURE 18**  
**EUTECTIC SALT SENSING ELEMENT**

The eutectic salt elements represent the minimum development risk approach but the highest unit weight. The high vapor pressure fluid (tube element contained) approach was selected over the others because of its potential overall capability and low unit weight.

The integration of the failure detection methods into an overall failure detection system is portrayed in Figure 19. The failure detection system includes instrumentation and crew displays for failure detection as well as for system monitoring.

The total cooled aircraft surface,  $2980 \text{ m}^2$  ( $32,134 \text{ ft}^2$ ), was divided into failure detection system control zones to illustrate how a detection system utilizing the fluid filled elements (phase change) would be configured. Figure 20 presents a schematic of the system. Each output signal from a control zone represents the sensor output signal from an individual panel. The signal from the panel sensors would be transmitted, by pressure transducers, to the local control zone micro-processor. This processor would, using memory, digital logic and computation, make decisions and issue commands to a central processor. The central processor monitors input data from all local micro-processors and provides the data to a video monitor, an audio monitor, and to a continuous recorder of panel temperature status. The use of semiconductor technology and large scale integration (LSI) devices for the local micro-processors and central control and display electronics will result in reliable low weight components.

A signal from any of the failure detection sensors indicating a failure condition is routed through system electronics to a crew display panel, resulting in illumination of the master light, an audio tone, and illumination of that particular parameter displayed on the panel. The system monitor sensors shown in Figure 19 detect the same parameters as the failure detection sensors plus other instrumentation considered to be of value for system monitoring. Outputs from these sensors provide a read-out of the parameter on the display panel. A discrete failure signal from either a failure detection sensor or its corresponding monitor sensor will alert the crew by master light illumination, an audio tone, and illumination of the failure detection parameter display. In-flight verification of the electronics and circuitry associated with each failure detection sensor would be provided by built-in-test (BIT) or press-to-test provisions.

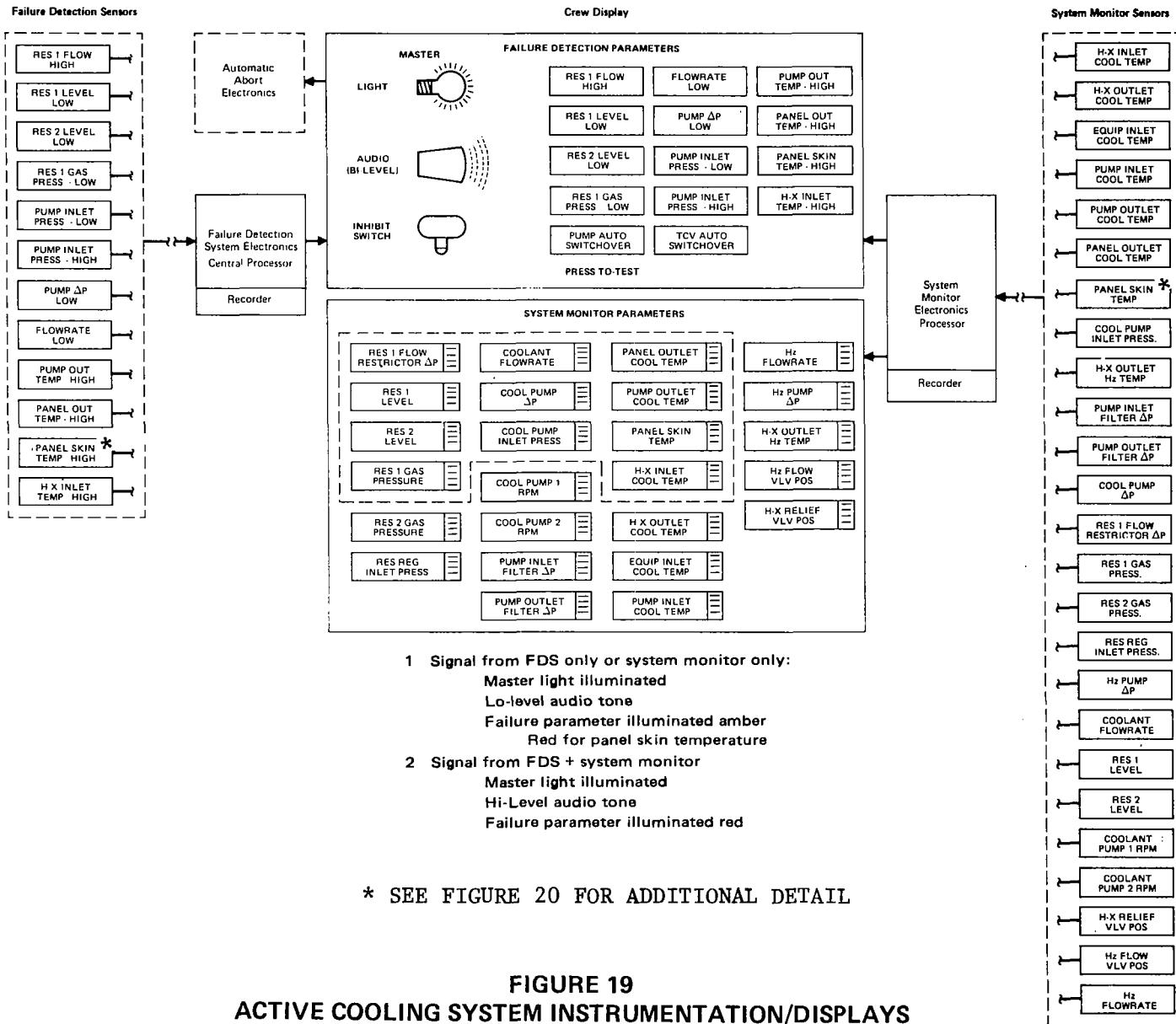
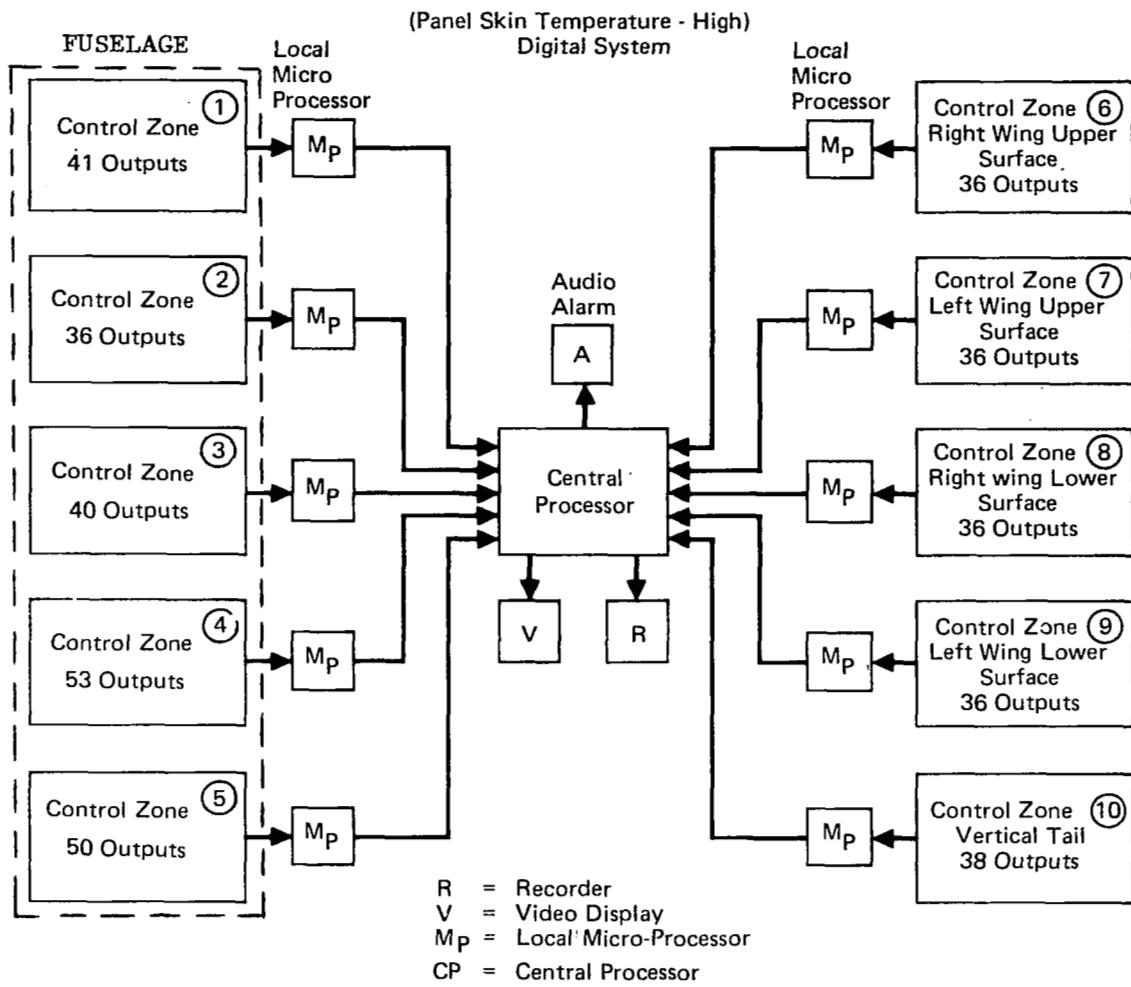


FIGURE 19  
ACTIVE COOLING SYSTEM INSTRUMENTATION/DISPLAYS



Each Control Zone Output Represents Output Signal for Individual Actively Cooled Panel

**FIGURE 20**  
**FAILURE DETECTION SYSTEM CONTROL AND DISPLAY ELECTRONICS**

Figure 19 also illustrates an output from the master panel to automatic abort mode electronics. In the case of a cooling system malfunction which is judged (by the central data processors, if crew reaction is not within an established time limit) to warrant abort, the automatic abort mode electronics would take command of the flight control system and initiate the abort maneuver.

A summary of system characteristics, viewed from the standpoint of response and reliability, is shown in Figure 21. Although some protection from false alarms is provided, further consideration of reliability might lead to quad-redundant sensors in certain critical locations.

- **Response**
    - Instrumentation Selected to Respond to Earliest Positive Failure Cues
    - Most Significant Failures Provide Multiple Cues Which are Also Detected
    - Visual and Audio Failure Signals Provided to Crew
    - Quantitative Response Times Dependent on Detailed System Characteristics and Specific Failure Modes
  
  - **Reliability**
    - Proven Reliable Instrumentation Used Where Possible
    - FDS Instrumentation Electronics and Circuitry Checked Periodically Before and During Sustained Cruise by Crew via Press-to-Test Capability
    - System Monitor Parameters Checked by Crew to Verify System Parameters are Within Tolerance
    - For Each Failure Parameter, a Cue is Available from Either the FDS Display or the System Monitor Display
      - ∴ A Double Failure Would Have to Occur to Prevent the Crew from Being Alerted to Out-of-Tolerance Condition for Each Parameter
    - One Failure Generally Results in Another Secondary Failure Cue Which is also Monitored by the Failure Detection System
      - ∴ Redundancy is also Provided by Detection of Secondary Failure Cues
    - Protection from False Alarms Provided by Requirement that Failure Signals from Both the FDS and the System Monitor Sensors Must Occur Before Failure Confirmation (Except for Skin Temperature)
- Failure Confirmation by:
- Hi-Level Audio Signal
  - FDS Display Illuminated Red

## FIGURE 21 FAILURE DETECTION SYSTEM DESIGN FEATURES/CONSIDERATIONS

### 4.3 ABORT TRAJECTORIES

Once a failure of the active cooling system, or an individual panel is detected, an abort maneuver is initiated. The manner in which the aircraft is maneuvered can result in significant effects on the total heat load that must be absorbed.

Detailed studies were made of abort from the selected cruise altitudes and Mach numbers. The limits of aircraft maneuvers during abort, such as maximum allowable "g" and maximum allowable angle of attack, were examined. The abort maneuvers were constrained to the allowable structural-aerodynamic-crew/passenger limitations.

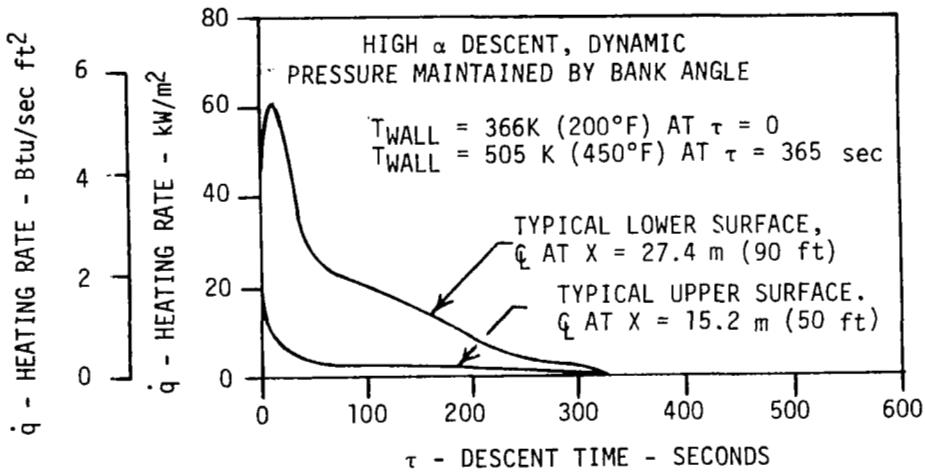
The abort trajectories for descent from Mach 3, 4.5, and 6 cruise utilized the following constraints. The maximum allowable load factor was 2.5 g's. The minimum flight dynamic pressure was limited to 4.79 kPa (100 lbf/ft<sup>2</sup>) to

insure adequate control at apogee. The maximum permissible angle of attack was set at 20 degrees for all abort Mach numbers. Aircraft designed to operate at these Mach numbers can be expected to have adequate control at 20 degrees angle of attack.

The most effective abort trajectories, in terms of reduced descent heat loads, were constant g-load pull-up (zoom) maneuvers. Bank angle (up to 80 degrees) was used to control apogee altitude. Figure 22 shows the heating rate histories for a trajectory of this type during descent from Mach 6 cruise.

#### START OF ABORT CONDITIONS

- o MACH 6, 32 km (105,000 ft),  $\alpha = 7^\circ$
- o AIRCRAFT WT. = 257.43 Mg (567,529 lbm)
- o ZERO THRUST
- o  $S_{REF} = 960 \text{ m}^2 (10,377 \text{ ft}^2)$



**FIGURE 22**  
MACH 6 ABORT TRAJECTORY HEATING RATES FOR  
AIRCRAFT TYPICAL LOWER AND UPPER SURFACES

Figure 23 presents the results of the abort heat load studies. These descent heat loads illustrate the magnitude of the amount of heat that the typical aircraft surface must absorb during a loss of active cooling abort from cruise conditions.

ABORT HEAT LOADS FOR TYPICAL AIRCRAFT SURFACES	START OF ABORT MACH NUMBER		
	3	4.5	6
LOWER SURFACE HEAT LOAD MJ/m <sup>2</sup> (Btu/ft <sup>2</sup> )	0.18 (15.9)	1.22 (107.5)	5.23 (460)
UPPER SURFACE HEAT LOAD MJ/m <sup>2</sup> (Btu/ft <sup>2</sup> )	0.073 (6.45)	0.20 (18)	0.74 (65)
AVERAGE* HEAT LOAD MJ/m <sup>2</sup> (Btu/ft <sup>2</sup> )	0.12 (10.9)	0.68 (60)	2.83 (249)

Note: \*The area weighted average heat load was based on the following:

$$Q_{avg} = \frac{A_L Q_L + A_U Q_U}{A_T} = 0.457 Q_L + 0.543 Q_U, \text{ and was considered as representative of average heat load to the aircraft.}$$

FIGURE 23  
ABORT DESCENT HEAT LOADS

Figure 24 presents a comparison of the reduction available in descent trajectory heat loads due to zoom type maneuvers. Results from this study and those of Reference (4) and (8) are shown. It can be seen that the reductions from normal maximum L/D descent heat loads are about the same.

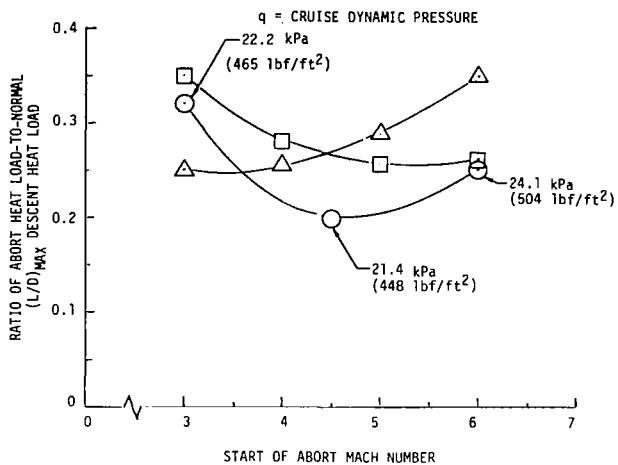
#### 4.4 FAIL-SAFE THERMOSTRUCTURAL CONCEPTS

A number of studies, References (7), (9), (10), (11), and (12), have shown that actively cooled structure has the potential for lower weight than hot structure for hydrogen fueled high speed aircraft. Because of its high thermal conductivity, high structural efficiency, and low cost, aluminum is presently the preferred material for actively cooled structure. However, in the event of inactivity following a cooling system failure, unprotected aluminum would rapidly overheat, resulting in catastrophic failure.

□ - REFERENCE (8) 2.0g PULL-UP FROM CRUISE  
AT  $q = 23.94 \text{ kPa}$  ( $500 \text{ lbf/ft}^2$ ),  $\alpha = 20^\circ$  GLIDE.

△ - REFERENCE (4) 2.0g PULL-UP FROM CRUISE AT  
 $q = 19.15 \text{ kPa}$  ( $400 \text{ lbf/ft}^2$ ),  $\alpha = 20^\circ$  GLIDE.

○ - THIS STUDY, REFINED TRAJECTORIES, 2.5g PULL-UP  
FROM CRUISE AT  $q$  SHOWN,  $\alpha = 20^\circ$  GLIDE, ABORT  $q$   
MINIMUM LIMITED TO  $4.79 \text{ kPa}$  ( $100 \text{ lbf/ft}^2$ )



**FIGURE 24**  
**COMPARISON OF ABORT MANEUVER HEAT LOADS**

Providing a redundant cooling system may not in all instances ensure against such a failure. Therefore, the objective of this portion of the study was to define and evaluate thermostructural concepts for actively cooled structure which will absorb an abort heat load in case of cooling system failure.

The basic concepts investigated included thickened outer skin, pre-cooled skin (a material used during normal operation at a temperature level well below its temperature limit to provide additional heat sink capacity), overcoats (including insulative metallic heat shields), undercoats, and phase-change materials (PCM).

A total of 44 combinations of the 3 Mach numbers, 2 aircraft surfaces and basic thermostructural concepts were selected for initial evaluation. Figure 25 illustrates this matrix.

A further screening and evaluation of the concepts resulted in selection of finalist concepts for each of the three abort Mach numbers. These finalist concepts were treated to more detailed analyses, to determine the characteristics of each, and final selections were made.

Thermo-Structural Concept	Cruise (Abort) Flight Condition/Panel Location					
	Mach 3 22.3 km (73,000 ft)		Mach 4.5 27.4 km (90,000 ft)		Mach 6 32 km (105,000 ft)	
	Upper Surface	Lower Surface	Upper Surface	Lower Surface	Upper Surface	Lower Surface
1. Thickened Skin a) Aluminum b) Boron-Alum. c) Lockalloy	● — —	● — —	● — —	● ● ●	● — —	● ● ●
2. Precooled Skin a) Aluminum b) Boron-Alum. c) Lockalloy	— — —	— — —	● — —	● ● ●	● — —	— ● ●
3. Overcoated Aluminum Skin a) Kapton b) Silicone c) Silicone (40% LiO <sub>2</sub> ) d) Insulative Heat Shield	● ● — —	● ● — —	● ● — —	● ● ● ●	● ● — —	● ● — ●
4. Undercoated Aluminum Skin a) Silicone (40% LiO <sub>2</sub> )	—	—	—	●	●	●
5. Phase Change Material (PCM) Aluminum Skin Alum. Honeycomb Polyethylene PCM	—	—	—	●	—	●
6. Baseline Aluminum Skin (Unprotected)	●	●	●	●	●	●

● Selected for further evaluation

**FIGURE 25**  
**MATRIX OF THERMOSTRUCTURAL CONCEPTS**

Figures 26, 27 and 28 present details of the panel design, coolant parameters, and maximum temperatures for the finalist candidates. The final selections are indicated in the figures by check marks.

Figure 29 presents a weight comparison of the selected concepts. In some cases the final selected configuration exhibited a slight weight penalty compared to the alternate concepts of Figures 26, 27 and 28. In these cases, the selected system resulted in a reduction in absorbed heat flux compared to the alternate concept and was selected to provide a more realistic match between absorbed heat flux and available heat sink.

Figure 30 presents the weights for the baseline bare unprotected actively cooled panels. These weights are included for comparative purposes. The weights shown are for the typical upper and lower surface locations with

AIRCRAFT SURFACE	THERMOSTRUCTURAL CONCEPT	ACTIVELY COOLED PANEL GEOMETRY						COOLANT PARAMETERS			MAX PANEL TEMPERATURE		
		P cm (in.)	D mm (in.)	t <sub>s</sub> mm (in.)	ΔX <sub>SIL</sub> mm (in.)	h cm (in.)	t <sub>i</sub> mm (in.)	HEATING RATE kJ/m <sup>2</sup> (Btu/sec ft <sup>2</sup> )	FLOW RATE PER TUBE g/s (lb/hr)	COOLANT ΔP kPa (psid)	MATERIAL	CRUISE TEMP K (°F)	ABORT TEMP K (°F)
UPPER	SILICONE OVERCOAT OVER AL. SKIN (HONEYCOMB PANEL)	10.6 (4.2)	6.35 (.25)	1.02 (.04)	1.17 (.046)	3.2 (1.26)	1.02 (.04)	4.17 (0.367)	10.84 (86)	22 (3.2)	AL. SKIN	383 (230)	386 (235)
	BARE AL. SKIN (HONEYCOMB PANEL)	10.6 (4.2)	6.35 (.25)	1.02 (.04)	-- --	3.2 (1.26)	1.02 (.04)	5.71 (0.503)	16.9 (134)	46 (6.7)	AL. SKIN	378 (221)	388 (239)
LOWER	SILICONE OVERCOAT OVER AL. SKIN (HONEYCOMB PANEL)	10.2 (4)	6.35 (.25)	1.02 (.04)	1.17 (.046)	3.07 (1.21)	0.97 (.038)	6.36 (0.56)	17.65 (140)	49 (7.1)	AL. SKIN	386 (235)	403 (266)
	BARE AL. SKIN (HONEYCOMB PANEL)	(2.95)	6.35 (.25)	1.02 (.04)	-- --	3.07 (1.21)	0.97 (.038)	11.55 (1.018)	23.8 (189)	81.4 (11.8)	AL. SKIN	386 (235)	416 (290)

$t_s$  = COOLED PANEL SKIN THICKNESS

$h$  = STRUCTURAL HONEYCOMB DEPTH

$P$  = COOLANT TUBE PITCH

$t_i$  = HONEYCOMB BACK-FACE SHEET THICKNESS

$D$  = COOLANT DEE TUBE DIAMETER

$\Delta X_{SIL}$  = SILICONE OVERCOAT THICKNESS

SELECTED CONFIGURATIONS

FIGURE 26  
MACH 3 THERMOSTRUCTURAL CONCEPTS

AIRCRAFT SURFACE	THERMOSTRUCTURAL CONCEPT	ACTIVELY COOLED PANEL GEOMETRY								COOLANT PARAMETERS			MAX PANEL TEMPERATURE		
		P cm (in.)	D mm (in.)	t <sub>s</sub> mm (in.)	ΔX <sub>SIL</sub> mm (in.)	h cm (in.)	t <sub>i</sub> mm (in.)	t <sub>o</sub> mm (in.)	ΔX <sub>INS</sub> mm (in.)	HEATING RATE kW/m <sup>2</sup> (Btu/sec ft <sup>2</sup> )	FLOW RATE PER TUBE g/s (lbm/hr)	COOLANT ΔP kPa (psid)	MATERIAL	CRUISE TEMP K (°F)	ABORT TEMP K (°F)
UPPER	SILICONE OVERCOAT OVER AL. SKIN	8.89 (3.5)	6.35 (.25)	1.02 (.04)	1.91 (.075)	3.1 (1.22)	0.91 (.036)	--	--	8.19 (0.722)	20 (159)	59 (8.6)	AL. SKIN	388 (239)	409 (276)
	BARE AL. SKIN	6.73 (2.65)	6.35 (.25)	1.02 (.04)	--	3.1 (1.22)	0.91 (.036)	--	--	14.44 (1.272)	26.4 (209)	95 (13.8)	AL. SKIN	533 (499)	537 (507)
LOWER	INSULATED HEAT SHIELD	10.2 (4)	6.35 (.25)	1.02 (.04)	--	3.15 (1.24)	1.07 (.042)	0.41 (.016)	6.35 (.25)	3.59 (0.316)	9.71 (.77)	20 (2.9)	AL. SKIN	380 (224)	420 (296)
	SILICONE OVERCOAT OVER AL. SKIN (SKIN/STRINGER)	8.89 (3.5)	6.35 (.25)	3.61 (.142)	1.3 (.051)	--	--	--	--	13.69 (1.206)	32.5 (258)	141 (20.4)	AL. SKIN	663 (734)	677 (759)

 $t_s$  = COOLED PANEL SKIN THICKNESS

P = COOLANT TUBE PITCH

D = COOLANT DEE TUBE DIAMETER

h = STRUCTURAL HONEYCOMB DEPTH

 $t_i$  = HONEYCOMB BACK FACE SHEET THICKNESS $\Delta X_{SIL}$  = SILICONE OVERCOAT THICKNESS $\Delta X_{INS}$  = INSULATION THICKNESS $t_o$  = TITANIUM HEAT SHIELD MATERIAL THICKNESS.

✓ SELECTED CONFIGURATIONS

**FIGURE 27**  
**MACH 4.5 THERMOSTRUCTURAL CONCEPTS**

AIRCRAFT SURFACE	THERMOSTRUCTURAL CONCEPT	P cm (in.)	D mm (in.)	ts mm (in.)	ACTIVELY COOLED PANEL GEOMETRY					COOLANT PARAMETERS			MAX PANEL TEMPERATURE		
					$\Delta X_{SIL}$ mm (in.)	h cm (in.)	$t_i$ mm (in.)	$t_o$ mm (in.)	$\Delta X_{ins}$ mm (in.)	HEATING RATE kW/m <sup>2</sup> (Btu/sec ft <sup>2</sup> )	FLOW RATE PER TUBE g/s (lb/hr)	COOLANT AP kPa (psid)	MATERIAL	CRUISE TEMP K (°F)	ABORT TEMP K (°F)
UPPER ✓	SILICONE OVERCOAT OVER AL. SKIN	7.47 (2.92)	6.35 <td>1.02 (.04)</td> <td>1.42 (.056)</td> <td>3.07 (1.21)</td> <td>0.91 (.036)</td> <td>--</td> <td>--</td> <td>12 (1.057)</td> <td>24.59 (195)</td> <td>85.5 (12.4)</td> <td>AL. SKIN</td> <td>391 (244)</td> <td>443 (337)</td>	1.02 (.04)	1.42 (.056)	3.07 (1.21)	0.91 (.036)	--	--	12 (1.057)	24.59 (195)	85.5 (12.4)	AL. SKIN	391 (244)	443 (337)
	BARE AL. SKIN	6.02 (2.37)	6.35 (.25)	1.02 (.04)	--	3.07 (1.21)	0.91 (.036)	--	--	17.64 (1.554)	29.5 (234)	121 (17.5)	AL. SKIN	391 (244)	454 (358)
LOWER ✓	INSULATED HEAT SHIELD	10.2 (4)	6.35 (.25)	1.02 (.04)	--	3.15 (1.24)	1.07 (.042)	0.41 (.016)	5.84 (.23)	4.26 (0.375)	11.73 (93)	26 (3.8)	AL. SKIN	379 (222)	465 (377)
													RENE' 41 HEAT SHIELD MATERIAL THICKNESS	850 (1070)	866 (1100)

ts = COOLED PANEL SKIN THICKNESS

P = COOLANT TUBE PITCH

D = COOLANT DEE TUBE DIAMETER

h = STRUCTURAL HONEYCOMB DEPTH

$t_i$  = HONEYCOMB BACK FACE SHEET THICKNESS

$\Delta X_{SIL}$  = SILICONE OVERCOAT THICKNESS

$\Delta X$  = INSULATION THICKNESS

$t_o$  = RENE' 41 HEAT SHIELD MATERIAL THICKNESS

✓ SELECTED CONFIGURATIONS

FIGURE 28  
MACH 6 THERMOSTRUCTURAL CONCEPTS

MACH NO.	AIRCRAFT SURFACE	CONFIGURATION (HONEYCOMB PANEL GEOMETRY)	FAIL-SAFE SYSTEM WEIGHT PER $7.43 \text{ m}^2$ ( $80 \text{ ft}^2$ ) PANEL ~ kg (lbm)						UNIT WEIGHT kg/m <sup>2</sup> (lbm/ft <sup>2</sup> )
			COOLED SKIN, TUBES, AND STRUCTURE	MANIFOLDS AND NON-OPTIMUMS	HEAT SHIELD AND INSULATION	ACTIVE COOLING SYSTEM 	FAILURE DETECTION SYSTEM 	TOTAL	
3	UPPER	SILICONE OVERCOAT OVER AL. SKIN	57.0 (125.6)	37.0 (81.6)	3.5 (7.7)	4.8 (10.6)	1.2 (2.6)	103.5 (228.1)	13.91 (2.85)
	LOWER	SILICONE OVERCOAT OVER AL. SKIN	57.0 (125.6)	37.0 (81.6)	3.5 (7.7)	6.6 (14.5)	1.3 (2.8)	105.4 (232.2)	14.16 (2.90)
4.5	UPPER	SILICONE OVERCOAT OVER AL. SKIN	57.0 (125.6)	37.0 (81.6)	5.7 (12.5)	8.1 (17.8)	1.5 (3.2)	109.3 (241.0)	14.70 (3.01)
	LOWER	INSULATED HEAT SHIELD OVER AL. SKIN	57.0 (125.6)	37.0 (81.6)	21.1 (46.5)	4.4 (9.6)	1.3 (2.8)	120.8 (266.1)	16.26 (3.33)
6	UPPER	SILICONE OVERCOAT OVER AL. SKIN	57.0 (125.6)	37.0 (81.6)	4.2 (9.3)	10.3 (22.8)	1.7 (3.8)	110.2 (243.1)	14.84 (3.04)
	LOWER	INSULATED HEAT SHIELD OVER AL. SKIN	57.0 (125.6)	37.0 (81.6)	40.6 (89.6)	4.9 (10.8)	1.3 (2.8)	140.8 (310.4)	18.94 (3.88)

NOTES:  SYSTEM HEAT LOAD IS EQUAL TO, OR LESS THAN, AVAILABLE HYDROGEN FUEL FLOW HEAT SINK CAPACITY.

 FLUID FILLED TUBE (PHASE CHANGE) FAILURE DETECTION ELEMENTS, WEIGHT BASED ON TUBE PITCH.  
SYSTEM WEIGHT IS  $0.168 \text{ kg/m}^2$  ( $0.0345 \text{ lbm/ft}^2$ ) FOR 10.4 cm (4 in.) TUBE PITCH. COOLING SYSTEM WEIGHT INCLUDES FAILURE DETECTION WEIGHT FOR BASIC COOLING SYSTEM.

FIGURE 29  
FAIL-SAFE SYSTEM WEIGHTS

MACH NO.	AIRCRAFT SURFACE	CONFIGURATION (HONEYCOMB PANEL GEOMETRY)	BASELINE WEIGHT PER 7.43 m <sup>2</sup> (80 ft <sup>2</sup> )					UNIT WEIGHT kg/m <sup>2</sup> (1bm/ft <sup>2</sup> )
			COOLED SKIN, TUBES, AND STRUCTURE	MANIFOLDS AND NON- OPTIMUMS	HEAT SHIELD AND INSULATION	ACTIVE COOLING SYSTEM	FAILURE DETECTION SYSTEM	
3	UPPER	BASELINE AL. SKIN AT AVERAGE Tw = 366K (200°F)	57.9 (127.7)	37.0 (81.6)	---	6.1 (13.5)	---	101.0 (222.8) 13.59 (2.79)
	LOWER	SAME AS ABOVE	58.3 (128.6)	37.0 (81.6)	---	10.1 (22.3)	---	105.4 (232.5) 14.19 (2.91)
4.5	UPPER	SAME AS ABOVE	59.0 (130)	37.0 (81.6)	---	13.2 (29.0)	---	109.2 (240.6) 14.70 (3.01)
	LOWER	SAME AS ABOVE	59.7 (131.7)	37.0 (81.6)	---	17.6 (38.9)	---	114.3 (252.2) 15.38 (3.15)
6	UPPER	SAME AS ABOVE	59.0 (130.1)	37.0 (81.6)	---	13.7 (30.2)	---	109.7 (241.9) 14.76 (3.02)
	LOWER	SAME AS ABOVE	60.7 (133.8)	37.0 (81.6)	---	26.0 (57.3)	---	123.7 (272.7) 16.65 (3.41)

NOTES: ACTIVE COOLING SYSTEM HEAT LOADS EXCEED AVAILABLE HYDROGEN FUEL FLOW HEAT SINK CAPACITY.

SYSTEMS DO NOT HAVE CAPABILITY FOR DETECTION OF INDIVIDUAL PANEL FAILURES.

FIGURE 30  
BASELINE SYSTEM WEIGHTS

heating rates, at cruise, equal to the average for the entire aircraft upper, or lower, surface, respectively. As shown by the figures, the unit weights are not radically different. The baseline systems do not have the capability to sense a loss of cooling within individual actively cooled panels, or the capability to survive a loss of cooling. In addition the baseline systems do not provide a reduction in absorbed heat flux to a level where basic engine fuel flow demands provide adequate heat sink.

The weights presented in Figures 29 and 30 were used to provide the system comparisons presented in Section 4.1. The overall system weights are based on an average value for the  $2980 \text{ m}^2$  ( $32,134 \text{ ft}^2$ ) cooled surface area of the aircraft. This is discussed in Section 5 herein.

## 5. DISCUSSION OF RESULTS

This section provides a limited discussion of the results presented in Section 4. Complete details of all study analyses and results are available in Reference (1).

### 5.1 SELECTED FAIL-SAFE SYSTEMS

The fail-safe systems selected for each of the three typical cruise Mach number conditions are discussed. The key issues are: failure detection, thermostructural concepts to provide reasonable structural temperatures during an abort, and matching of absorbed heat flux with the available heat sink.

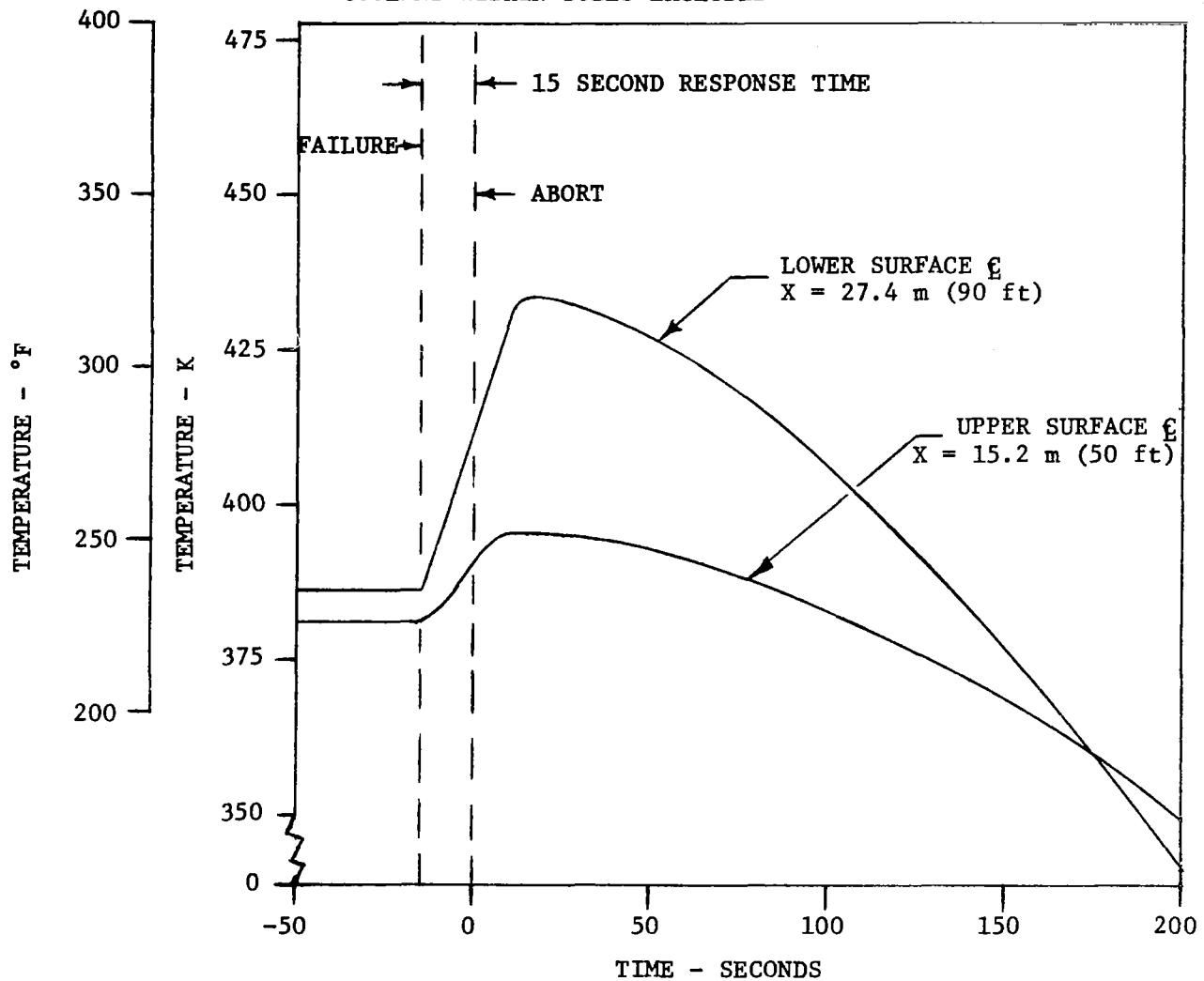
o Mach 3 Fail-Safe System - As shown in Section 4.1, the fail-safe system as configured for Mach 3 cruise used silicone elastomer overcoats on both the aircraft upper and lower surfaces. The silicone overcoats were selected primarily to provide a reduction in cruise heat transfer to the actively cooled structure. This is illustrated by a comparison of Figures 31 and 32, and by Figure 33.

Figure 31 shows the transient temperature characteristics of the bare aluminum panels (baseline) for Mach 3 abort. The maximum temperatures experienced by the aluminum structure remain below the reuse limit of 450 K (350°F). Figure 32 shows the transient temperatures for the silicone overcoated panels during abort. The temperatures of the aluminum structure are below those for the bare panels. However, the reductions in temperature are not enough to justify selection of the silicone overcoat rather than the bare panel designs. The key to the selection of the silicone was its insulative characteristics which reduced absorbed heat flux.

The impact of the addition of the silicone overcoat is illustrated by Figure 33. As shown, increasing the overcoat thickness (which increases external surface temperature and heat rejection by radiation) does not result in any measurable weight penalty until thicknesses in excess of the selected value are added. The absorbed heat flux, for the selected thickness, is equal to the available heat sink capacity of engine fuel flow.

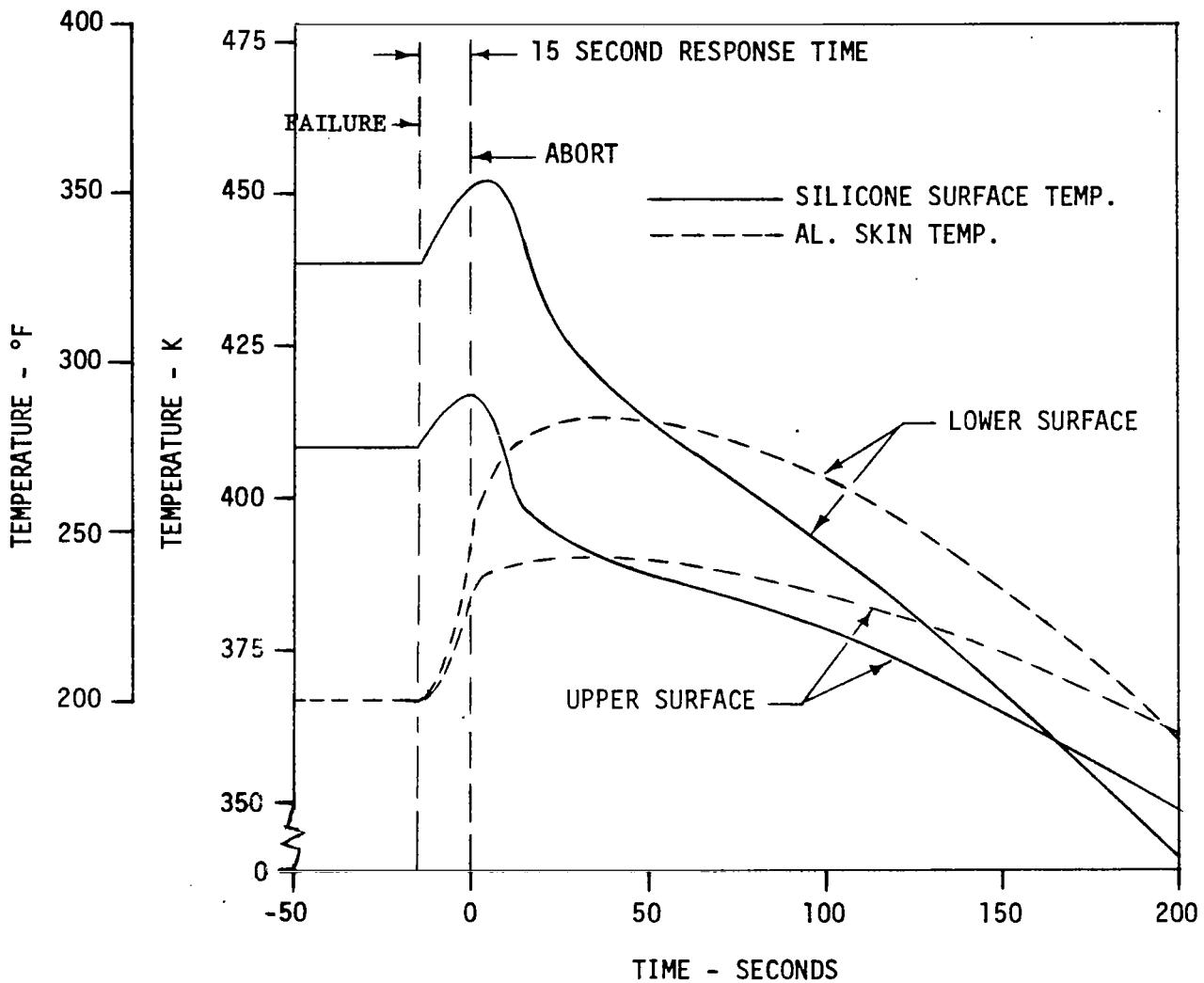
The estimate of available heat sink capacity is based on the use of duct burning turbofan engines for Mach 3 cruise. Data on typical fuel flow rates per square foot of aircraft wetted area were obtained from Reference (13).

- o 1.02 mm (0.04 in.) AL. PANEL
  - o HEAT CAPACITY OF PANEL SUPPORT STRUCTURE AND COOLANT WITHIN TUBES EXCLUDED



**FIGURE 31**  
**BASELINE PANEL SKIN TEMPERATURES DURING MACH 3 ABORT**

- SILICONE OVERCOAT THICKNESS = 1.17 mm (0.046 in.)
- ALUMINUM SKIN THICKNESS = 1.02 mm (0.04 in.)
- HEAT CAPACITY OF PANEL SUPPORT STRUCTURE AND COOLANT WITHIN TUBES EXCLUDED



**FIGURE 32**  
**SILICONE OVERCOAT CONCEPT ABORT TEMPERATURE RESPONSE**  
**(MACH 3 UPPER AND LOWER SURFACES)**

- 1) TYPICAL UPPER & LOWER PANELS
  - o PANEL SIZE = 1.2 m x 6.1 m (4 ft x 20 ft)
  - o 1.02 mm (0.04 in.) AL. SKIN
  - o PANEL COOLANT INLET TEMP = 255K (0°F); OUTLET = 322K (120°F)
  - o SYSTEM  $\Delta P$  = 1.17 MPa (170 psid)
- 2) PANEL WEIGHTS EXCLUDE SUPPORT STRUCTURE

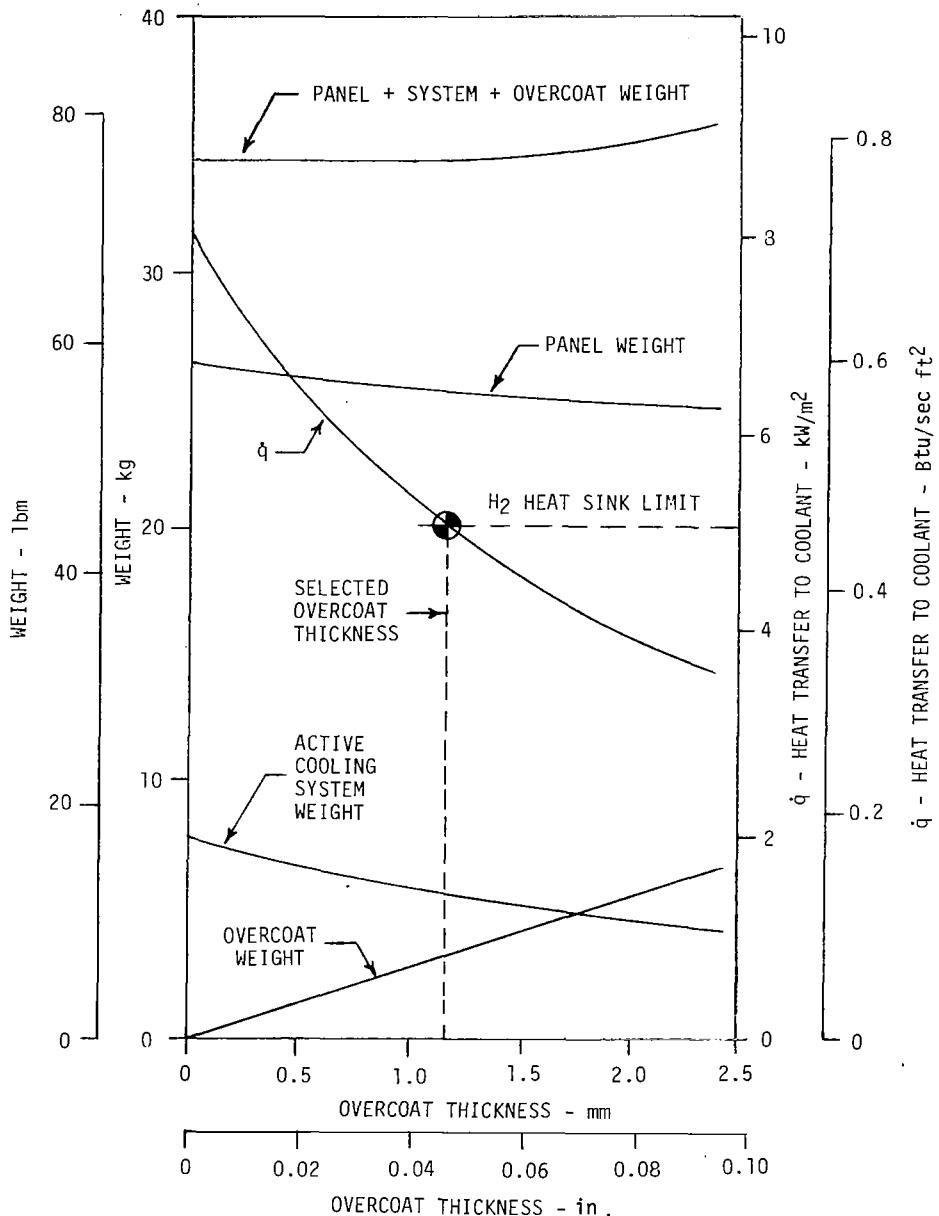


FIGURE 33  
MACH 3 CRUISE - SILICONE OVERCOAT

Temperature rise of the hydrogen fuel, upon absorbing the heat, was limited to a 278 K (500°F) increase. Thus, the hydrogen limit temperature was approximately 311 K (100°F).

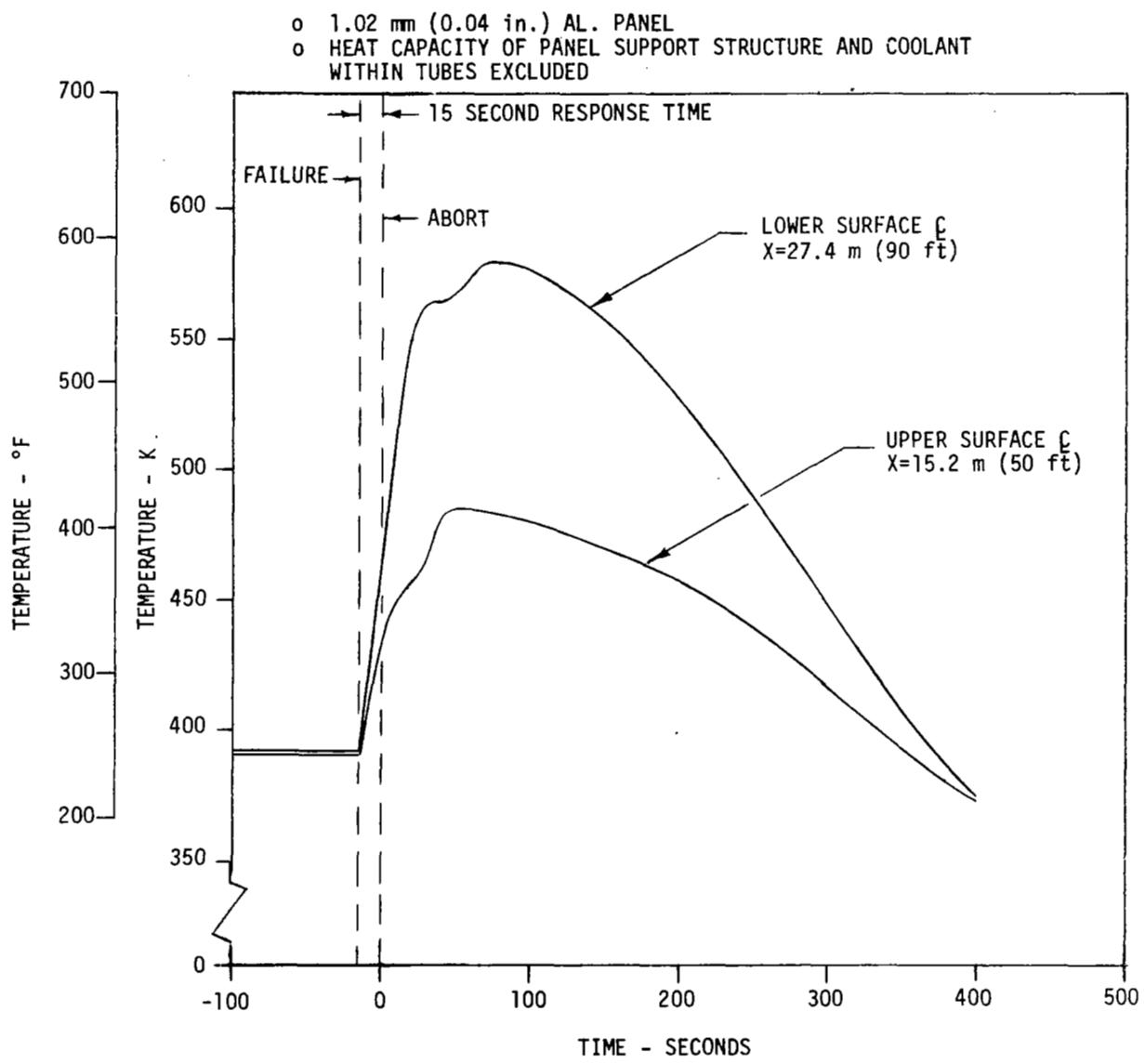
An analysis of the panel overtemperature detection elements indicated that a panel overtemperature condition (complete loss of cooling in one tube) could be sensed in less than 15 seconds with the sensor located adjacent to the coolant tubes.

o Mach 4.5 Fail-Safe System - Figure 34 shows transient temperature characteristics of the bare baseline aluminum panels during an abort mode descent. Maximum abort temperatures are well above the reuse limit of 450 K (350°F). Without detection of the loss of cooling, and thus, no abort, the typical upper surface panel would reach a steady state temperature of about 561 K (550°F). The lower surface, under these conditions, would reach a temperature level of about 663 K (734°F). Under these conditions, and especially if the aircraft were to pull load factors higher than normal cruise load factor, structural failure could take place.

Figures 35 and 36 show the temperatures experienced by the selected fail-safe concepts during abort. As shown, the temperatures of the structure remain below the reuse limit of 450 K (350°F). The insulation used for the lower surface design was a fibrous glass felt with a density of 64.1 kg/m<sup>3</sup> (4 lbm/ft<sup>3</sup>) and a 922 K (1200°F) continuous use temperature. The insulation packages were assumed bonded directly to the aluminum skin of the panel.

The selected fail-safe system, as described in Section 4.1, only requires about 41% of the available fuel heat sink capacity. The availability of heat sink was estimated from a correlation of typical hydrogen fuel flow per square foot of wetted area based on typical aircraft configurations and advanced turboramjet engines. Figure 37 presents this correlation. The available temperature rise of the hydrogen fuel was assumed limited with a maximum temperature of 311 K (100°F) after absorbing the aircraft heat load.

The fluid filled tube elements, used to provide a warning of structural overtemperature conditions, would be located near the coolant tubes, for the selected thermostructural concepts. As shown by Figures 35 and 36, the aluminum skin over the coolant tubes experience the greatest increase in temperature after failure and, thus, this location would result in the fastest sensor



**FIGURE 34**  
**BASELINE PANEL SKIN TEMPERATURES DURING MACH 4.5 ABORT**

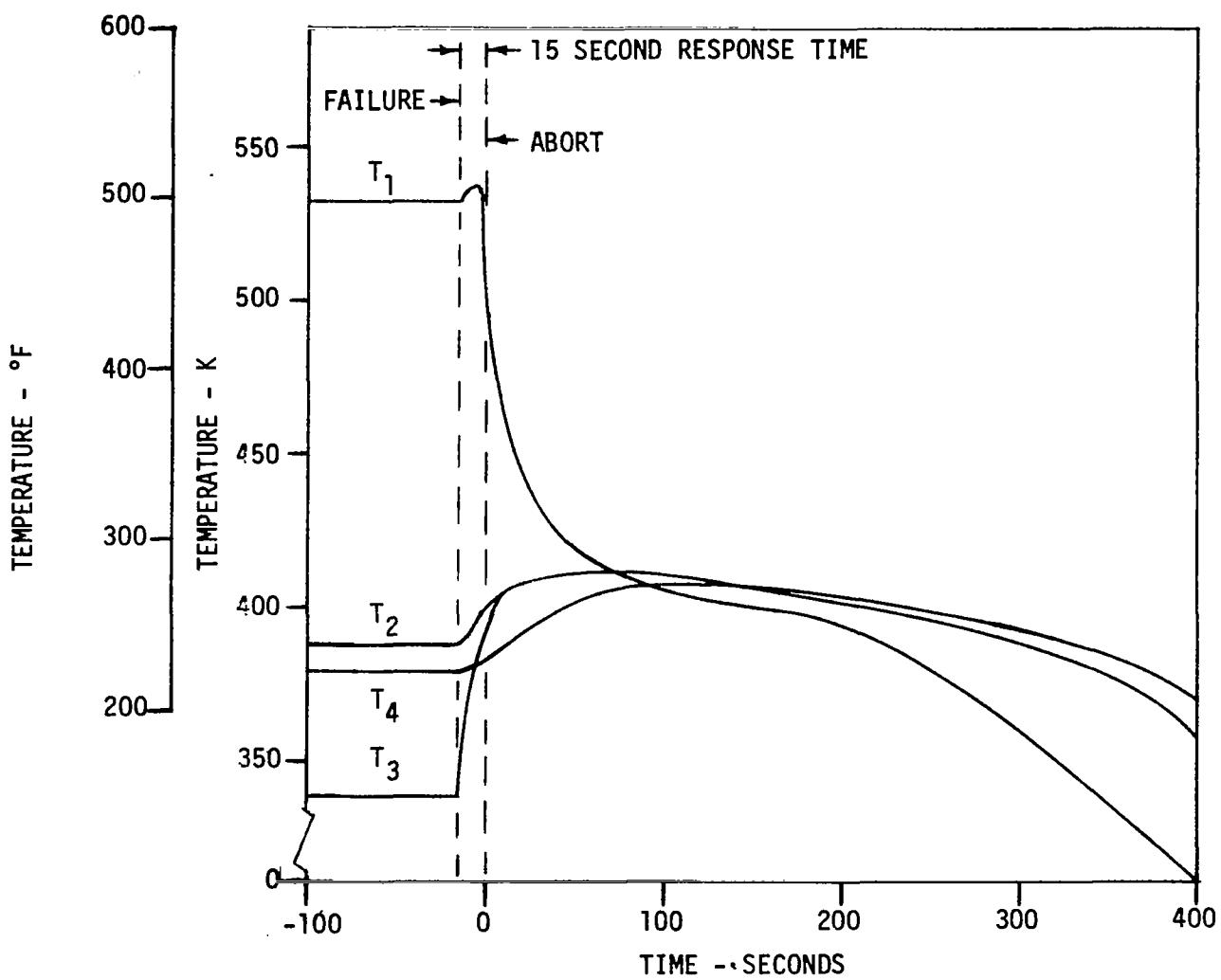
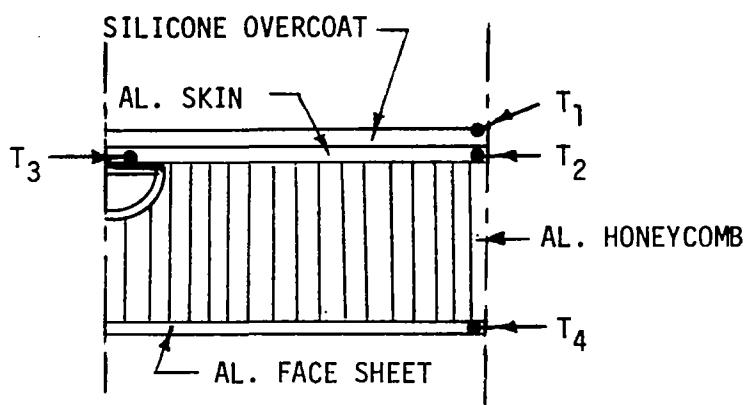
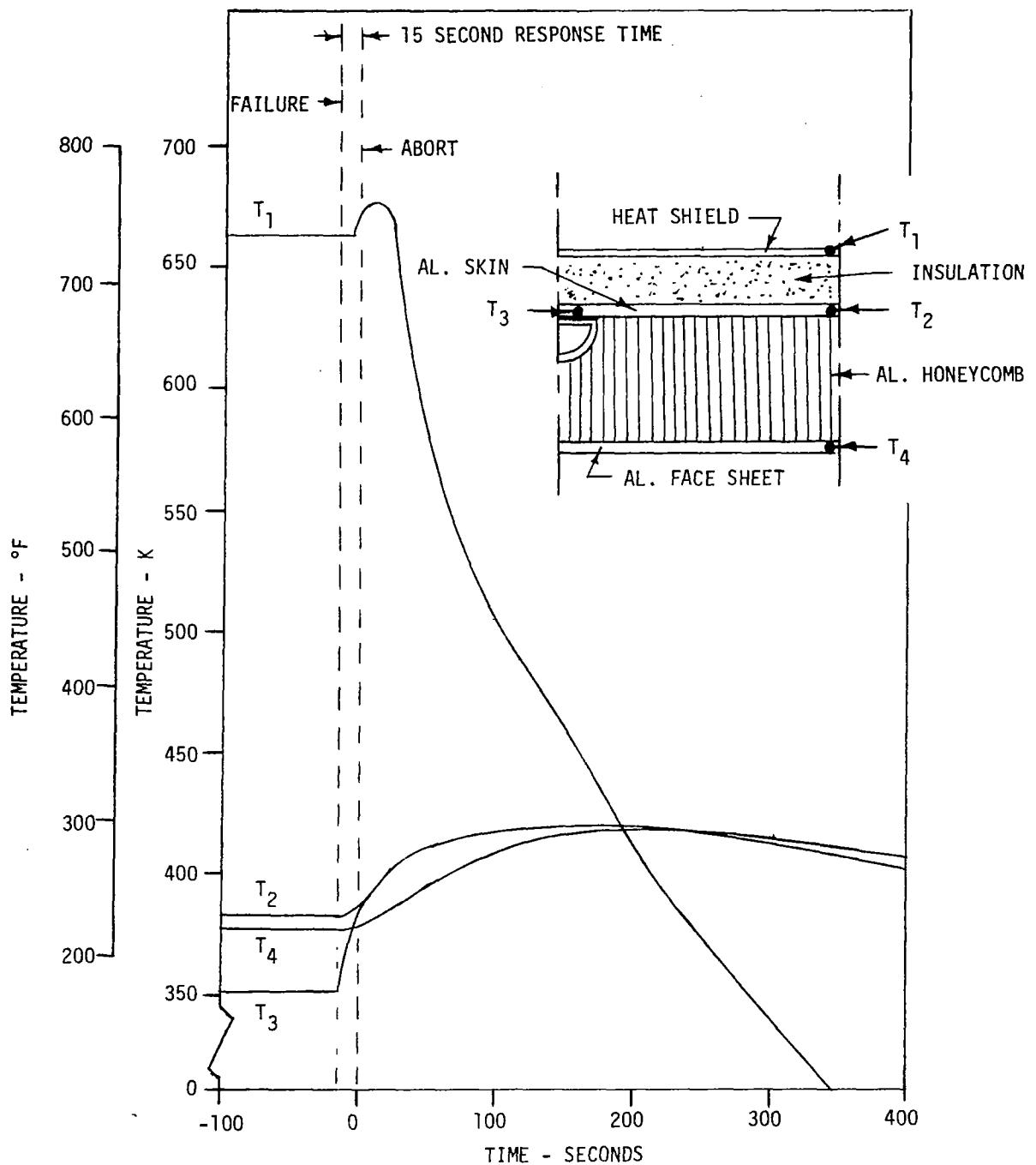


FIGURE 35  
MACH 4.5 UPPER SURFACE TEMPERATURES DURING ABORT  
(SILICONE OVERCOAT CONCEPT)



**FIGURE 36**  
**MACH 4.5 LOWER SURFACE TEMPERATURES DURING ABORT**  
**(INSULATIVE HEAT SHIELD CONCEPT)**

<u>SYMBOL</u>	<u>DATA SOURCE</u>	<u>ENGINE TYPE</u>
○	CURRENT STUDY	TURBO-RAMJET
□	REFERENCE 14	TURBO-RAMJET
△	REFERENCE 7	DUCT BURNING TURBOFAN
▲	REFERENCE 3	TURBO-RAMJET
▷	REFERENCE 15	SCRAMJET

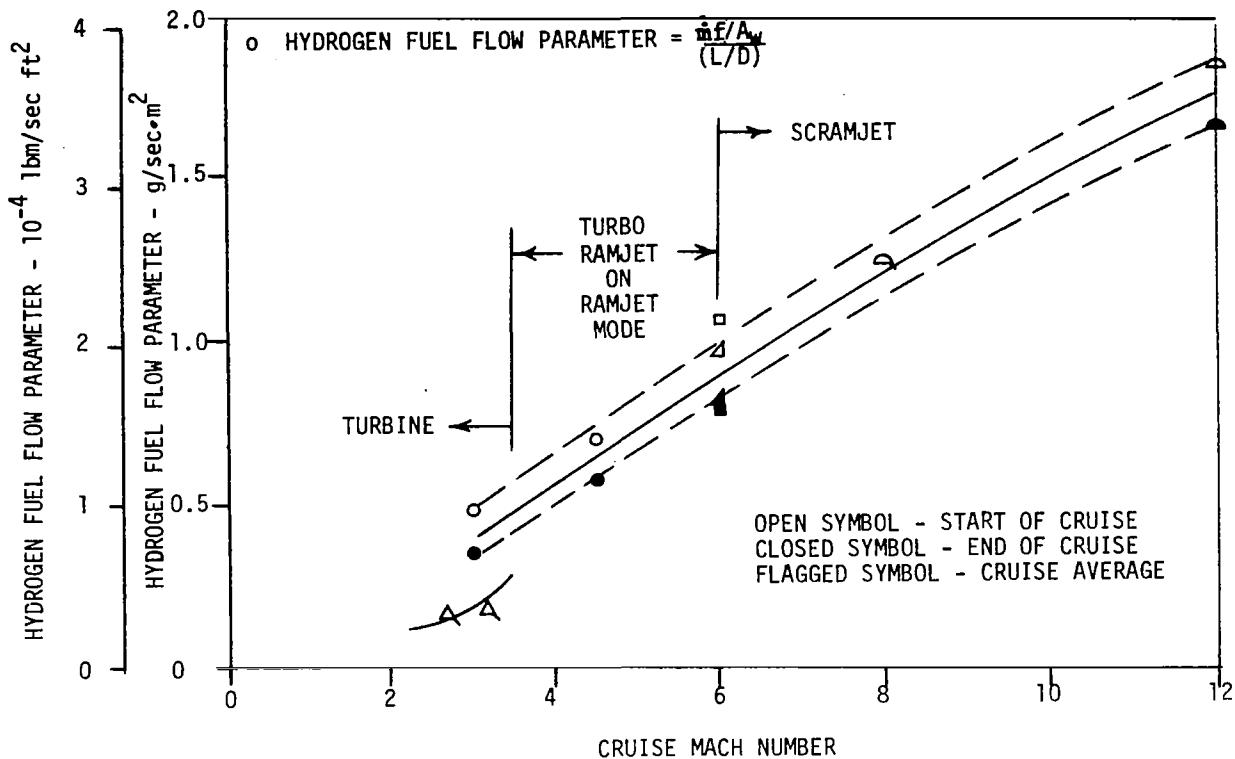
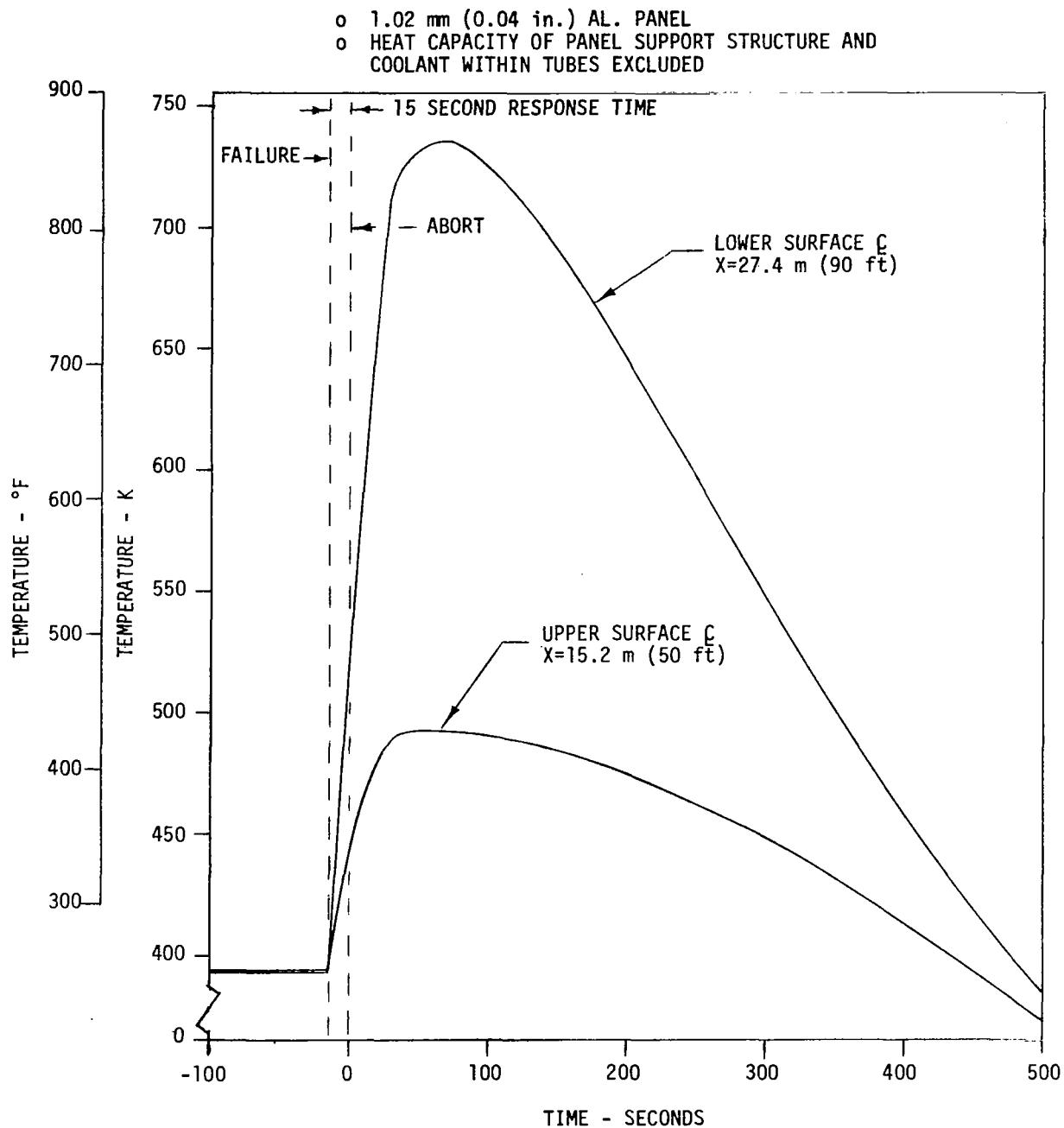


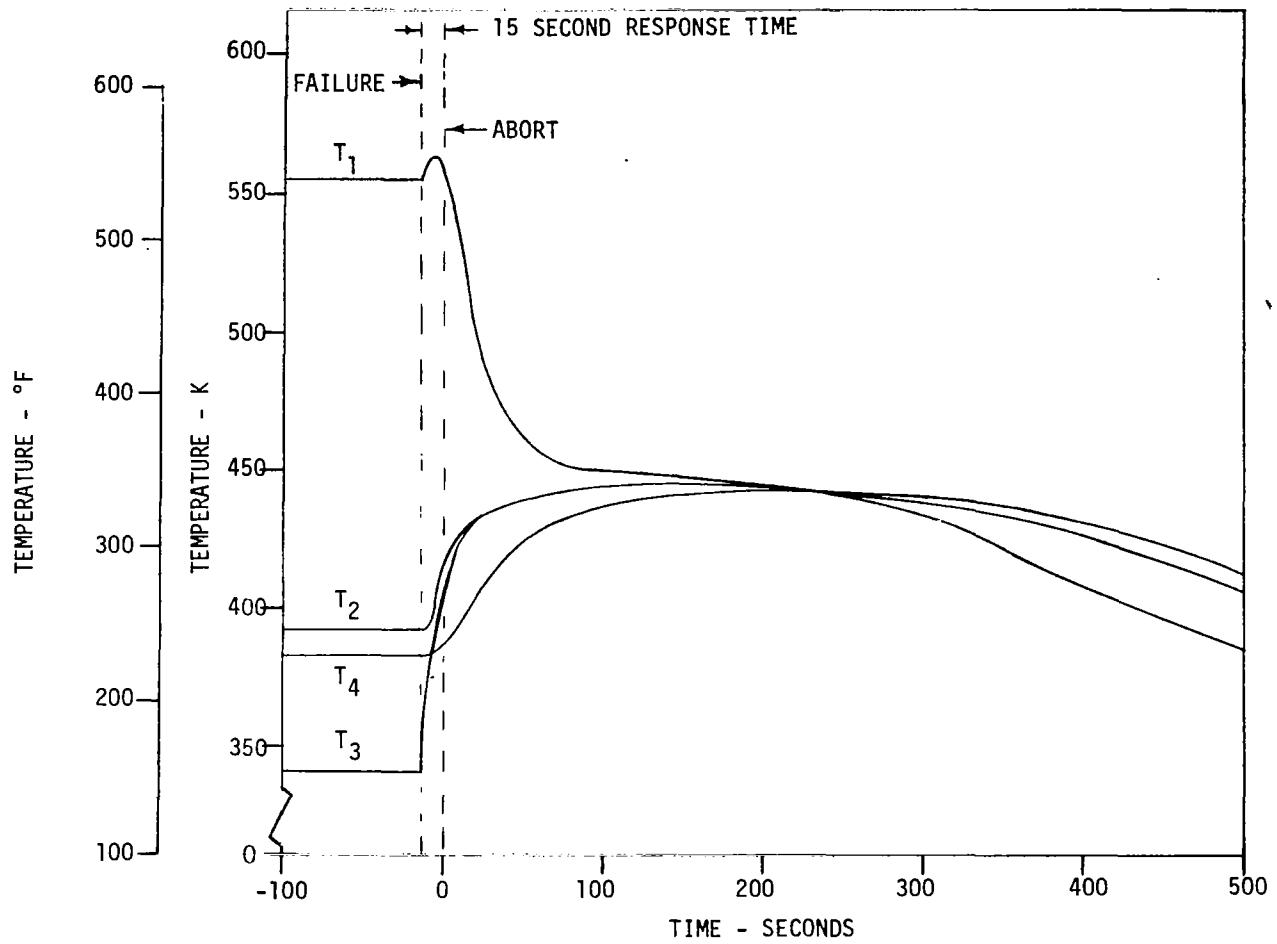
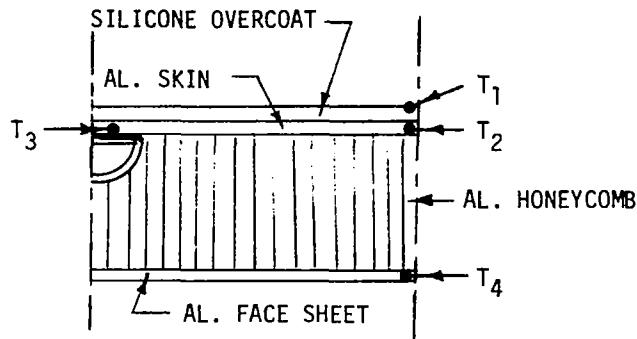
FIGURE 37  
HYDROGEN FUEL FLOW PARAMETER FOR HIGH  
MACH NUMBER CRUISE AIRCRAFT

response time. Based on an analysis of sensor characteristics it is judged that sensor response would be within the 15 second lag between failure and abort used in thermal analyses of the panel concepts.

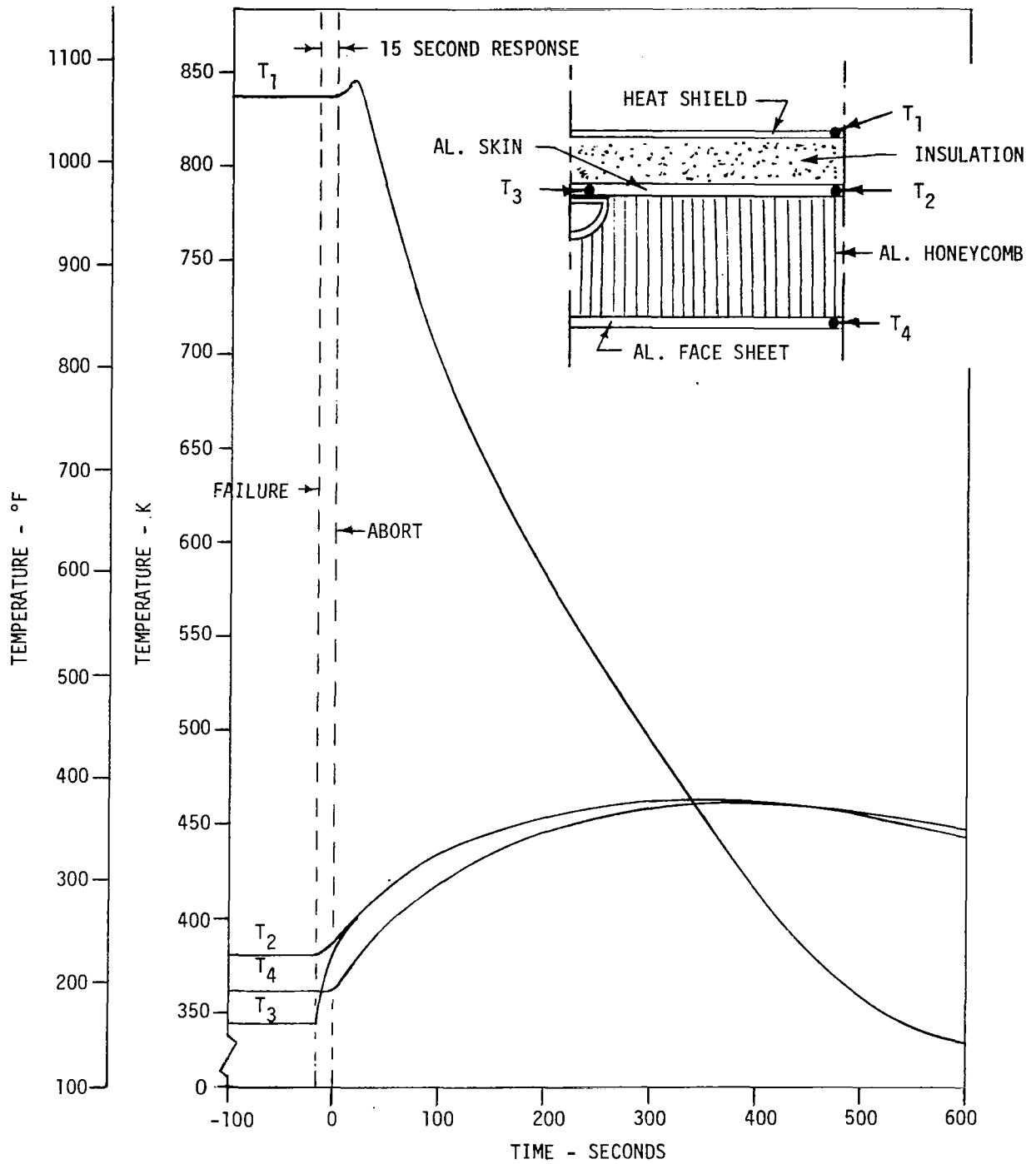
o Mach 6 Fail-Safe System - Figures 38, 39 and 40 are presented to illustrate the transient temperature characteristics experienced during an abort from Mach 6 cruise. Figure 38 shows the temperature response of a bare aluminum panel. The temperatures, if the loss in cooling is detected and abort initiated, reach about 736 K (865°F) on the aircraft lower surface and



**FIGURE 38**  
**BASELINE PANEL TEMPERATURES DURING MACH 6 ABORT**



**FIGURE 39**  
**MACH 6 UPPER SURFACE TEMPERATURES DURING ABORT**  
**(SILICONE OVERCOAT CONCEPT)**



**FIGURE 40**  
**MACH 6 LOWER SURFACE TEMPERATURES DURING ABORT**  
**(INSULATIVE HEAT SHIELD CONCEPT)**

about 491 K (425°F) on the upper surface. If undetected, the loss of cooling could result in temperatures of nearly 922 K (1200°F) on the lower surface and 589 K (600°F) on the upper surface.

Figures 39 and 40 show the temperature response of the selected Mach 6 thermostructural concepts for the upper surface and lower surface, respectively. Peak temperatures are moderate during the abort. The upper surface structure reaches 443 K (337°F) and the lower surface structure reaches 465 K (377°F), for the 15 second response time.

The thermostructural concepts selected for Mach 6 cruise reduce the absorbed heat load to only about 47% of available fuel heat sink capacity. Again, the availability of heat sink was derived from the Figure 37 correlation.

The Freon filled tube elements (located adjacent to the coolant tubes) used to provide actively cooled panel temperature monitoring, can detect the loss of an individual coolant tube cooling function within about 5 seconds after failure. The same element, if located at the midpoint between tubes, requires about 13 seconds to signal the loss of a tube function. These sensors, located next to the lower surface coolant tubes, could provide a warning of loss of tube cooling function in about 11 seconds.

## 5.2 FAILURE DETECTION

Figure 41 presents candidate failure detection methods for the failure modes identified for the active cooling system. Failure cues are shown in a manner to indicate the sequence in which they would be expected to occur, thus suggesting primary failure detection methods which allow the most time for reaction.

Conventional instrumentation (i.e., pressure and temperature sensors, flowmeters, etc.) can satisfactorily be employed for detecting most system failures. However, detection of flow restriction of individual tubes within the actively cooled panels by these methods presents significant practical problems because of the large number (over 400) of panels.

However, this type of failure would essentially have no effect on any other measurable system parameters (i.e., coolant flowrate,  $\Delta P$ , or temperatures of the overall coolant system or even of an individual panel). The failure could be isolated to one panel, or even to a part of a panel, without producing a discernible effect on the remainder of the system.

Failure Modes	Failure Detection Methods												
	Reservoir No. 1 Coolant Flow - High	Reservoir No. 1 Level - Low	Reservoir No. 2 Level - Low	Reservoir Gas Pressure - Low	Pump Inlet Pressure - Low	Pump Inlet Pressure - High	Pump ΔP - Low	Coolant Flowrate - Low	Pump Outlet Coolant Temp - High	Panel Outlet Coolant Temp - High	Panel Skin Temp - High	H-X Inlet Coolant Temp - High	Coolant Filter ΔP - High
1. Active Cooled Panel A. Leakage B. Restricted Tube	1 ● 2 ● 3 X				4 X		5 X 6 X		7 X 8 X 9 X				
2. Distribution Lines A. Leakage B. Restricted Branch Line C. Restricted Main Line	1 ● 2 ● 3 X				4 X	5 X 6 X	1 ●	7 X 8 X 9 X	1 ● 2 ● 3 X 4 ●				
3. Coolant Pump A. External Leakage B. Internal Leakage C. Low Flowrate ▲	1 ● 2 ● 3 X			4 X	5 X 6 X	1 ● 2 ●	7 X 8 X 9 X	1 ● 2 ● 3 X 4 ●					
4. Heat Exchanger A. External Leakage B. Internal Leakage C. Flow Restriction	1 ● 2 ● 3 X			4 X	5 X 6 X	1 ● 2 ● 3 ●	7 X 8 X 9 X	1 ● 2 ● 3 X 4 ●					
5. Temp Control Valve ▲ A. Failed Cold B. Failed Hot					1 ●	2 X 3 X	1 ● 2 ● 3 ● 4 ●	1 ● 2 ● 3 X 4 ●	1 ● 2 ● 3 X 4 ●				
6. Coolant Reservoir A. Coolant Leak B. Internal Gas Leak C. Loss of Gas Pressure	1 ● 2 ● 3 X		1 ● 2 ●	4 X	5 X 6 X	1 ● 2 ● 3 ● 4 ●	7 X 8 X 9 X	1 ● 2 ● 3 X 4 ●					
7. Coolant Filter A. Leakage B. Flow Restriction	1 ● 2 ● 3 X			4 X	5 X 6 X	1 ● 2 ● 3 ● 4 ●	7 X 8 X 9 X	1 ● 2 ● 3 X 4 ●	1 ●				
8. H-X Relief Valve A. External Leakage B. Internal Leakage	1 ● 2 ● 3 X			4 X	5 X 6 X	1 ● 2 ● 3 ● 4 ●	7 X 8 X 9 X	1 ● 2 ● 3 X 4 ●					
9. Ovbd Relief Valve ▲ A. Leakage	1 ● 2 ● 3 X			4 X	5 X 6 X	1 ● 2 ● 3 ● 4 ●	7 X 8 X 9 X						

 Candidate failure detection methods (N = sequence of failure cues)  
 Primary failure detection methods  
 Depicts failure modes backed up by redundancy

FIGURE 41  
METHODS OF ACTIVE COOLING SYSTEM FAILURE DETECTION

Preflight detection of clogged tubes might be achieved by visual inspection for dents or use of some method such as infrared scanning for identifying internal restrictions. However, detecting a restricted coolant tube, or a debonding of the tube from the skin, requires sensing the panel local skin temperature either directly or indirectly.

One of the study guidelines was to consider detection of local "hot spots" a primary concern. The problem requires both fast response time and the capability to survey a large surface area with a small number of sensors. Sensors in contact with the panel skin are subjected to all of the practical problems associated with the use of more conventional temperature sensors. A vast network would be required and would result in a highly complex system. Therefore, a list of schemes was expanded to include as many approaches as possible. These approaches included thermistors, thermocouples, thermal fuses, bimetal actuators, thin-film capacitors, thin-film resistors, paired transistors, eutectic salt temperature sensitive elements, fusible metal alloy electrical circuit grids, sealed tube elements containing fluids, infrared scanning, thermographic phosphors, temperature sensitive paints, release of a tracer material from temperature sensitive surface coatings which seed the external boundary layer, release of a tracer material within coolant tubes, etc. These approaches were given two levels of screening and the surviving candidates evaluated on the basis of cost, reliability, relative weight, and response time.

These surviving candidates were:

- o Eutectic salt elements (resistance change)
- o Fluid filled tube elements (thermal expansion)
- o Fluid filled tube elements (phase change)
- o Infrared scanning (surface IR emission)

On the basis of the evaluation and additional examination of compatibility with selected thermostructural concepts, the field was narrowed to two finalist approaches. These were the eutectic salt elements (electrical resistance change) and Freon filled tube elements (phase change).

Both types of elements can provide a signal upon loss of individual panel cooling in less time than the 15 second delay used in abort thermal analyses.

The eutectic salt elements would result in a penalty (fixed weight increase) of 1.79 Mg (3952 lbm) to the aircraft based on an average tube pitch of 10.16 cm (4 in.). The Freon filled tube elements would result in a penalty of 0.5 Mg (1109 lbm).

Thus, the Freon filled tubes would be superior to the eutectic salt elements. In addition, the eutectic salt element can only provide failure indication whereas the Freon tubes provide a continuous monitoring capability.

The eutectic salt elements are state-of-the-art. The Freon filled tube element approach, to our knowledge, has never been considered for application as a failure detection device and would require considerable development. The risk appears low insofar as the engineering technology is concerned. The major cost/risk factor would be in the fabrication and installation of a reliable functioning system. The impact on panel fabrication cost (assuming a fully developed tube system) is estimated to be approximately the same for either approach. It may prove that some fluid other than Freon will be more desirable in this application. However, Freon illustrates the basic principal of operation. Determination of the optimum fluid to use is considered as part of the required development. The conclusion was that the potential payoff of the Freon filled sensors, a safe aircraft system, appears to justify the selection of this approach for integration into advanced technology aircraft.

### 5.3 ABORT TRAJECTORIES

The most nebulous of the trajectory constraints used in the studies were the physiological limitations of the crew and passengers. These limitations were influenced significantly by discussions with airline pilots. After hearing an explanation of why abort maneuvers are a requirement for the "Fail-Safe Abort System" aircraft, three out of three pilots related these maneuver to collision avoidance maneuvers. In all cases the feeling was that the crew would "pull the wings off" to avoid a mid-air collision. The abort from high speed cruise was placed in the same category. The aircraft would be maneuvered to its maximum "g-load" capability. Passenger safety during the maneuver would be a major concern. However, the prime concern is to save the majority.

The maximum allowable inflight load factors defined for the baseline aircraft (Reference (5)) were used in final trajectory studies. The baseline aircraft was designed for a limit load factor ( $n_z$ ) of +2.5, -1.0 (Z direction

positive upward) at the normal structural panel operating temperatures of 366 K (200°F) average, 394 K (250°F) maximum. Ultimate shear and bending moment curves for the baseline aircraft are available in Reference (5).

For an abort condition, Reference (15) specifies that the structure must be capable of carrying 80% of the design limit loads. All of the final selected panel designs are capable of carrying more than 100% design limit load at 478 K (400°F).

A total of 15 descent trajectories were investigated; 8 for Mach 6, 3 for Mach 4.5, and 4 for Mach 3. One of these for each Mach number was descent at maximum L/D to provide a reference descent heat load. In all cases, the most effective abort descent to reduce overall aircraft heat loads was found to be zoom type maneuvers. These maneuvers used a maximum g-load pull-up to the maximum angle of attack (20°) glide condition. Bank angle was used to control apogee altitude (minimum dynamic pressure of 4.79 kPa (100 lbf/ft<sup>2</sup>)). This insured adequate aerodynamic control at apogee.

Drag brakes were investigated as a means of reducing L/D, and thus descent time, at Mach 6. However, the 70.7 m<sup>2</sup> (761 ft<sup>2</sup>) of split rudder opened at 45° flare, utilized as a drag brake, was not effective at the higher angles of attack. The kinetic energy of the aircraft at Mach 6 cruise completely overpowers the available drag brake area. It is highly doubtful if it is practical to install a large enough drag brake area on an aircraft of this size to make any significant difference in descent time.

Figure 42 provides a comparison of a maximum L/D descent and a zoom type abort descent from Mach 6 cruise. The angle of attack during various descent time steps was held constant, except for the zoom. This descent mode results in a phugoid oscillation. The oscillation could be prevented by modulating the angle of attack so that the equilibrium altitude corresponding to the lift coefficient is maintained. However, this type of trajectory does not reduce the Mach number as rapidly as the oscillating trajectory. The maximum L/D descent resulted in an average heat load of 11.08 MJ/m<sup>2</sup> (975 Btu/ft<sup>2</sup>). The zoom maneuver resulted in an average of 2.83 MJ/m<sup>2</sup> (249 Btu/ft<sup>2</sup>). These results are consistent, in terms of relative descent heat loads, with the results of Reference (8).

START OF ABORT CONDITIONS

- o MACH 6, 32 km (105,000 ft),  $\alpha = 7^\circ$
- o AIRCRAFT WT. = 257.43 Mg (567,529 lbm)
- o ZERO THRUST
- o S<sub>REF</sub> = 960 m<sup>2</sup> (10,377 ft<sup>2</sup>)

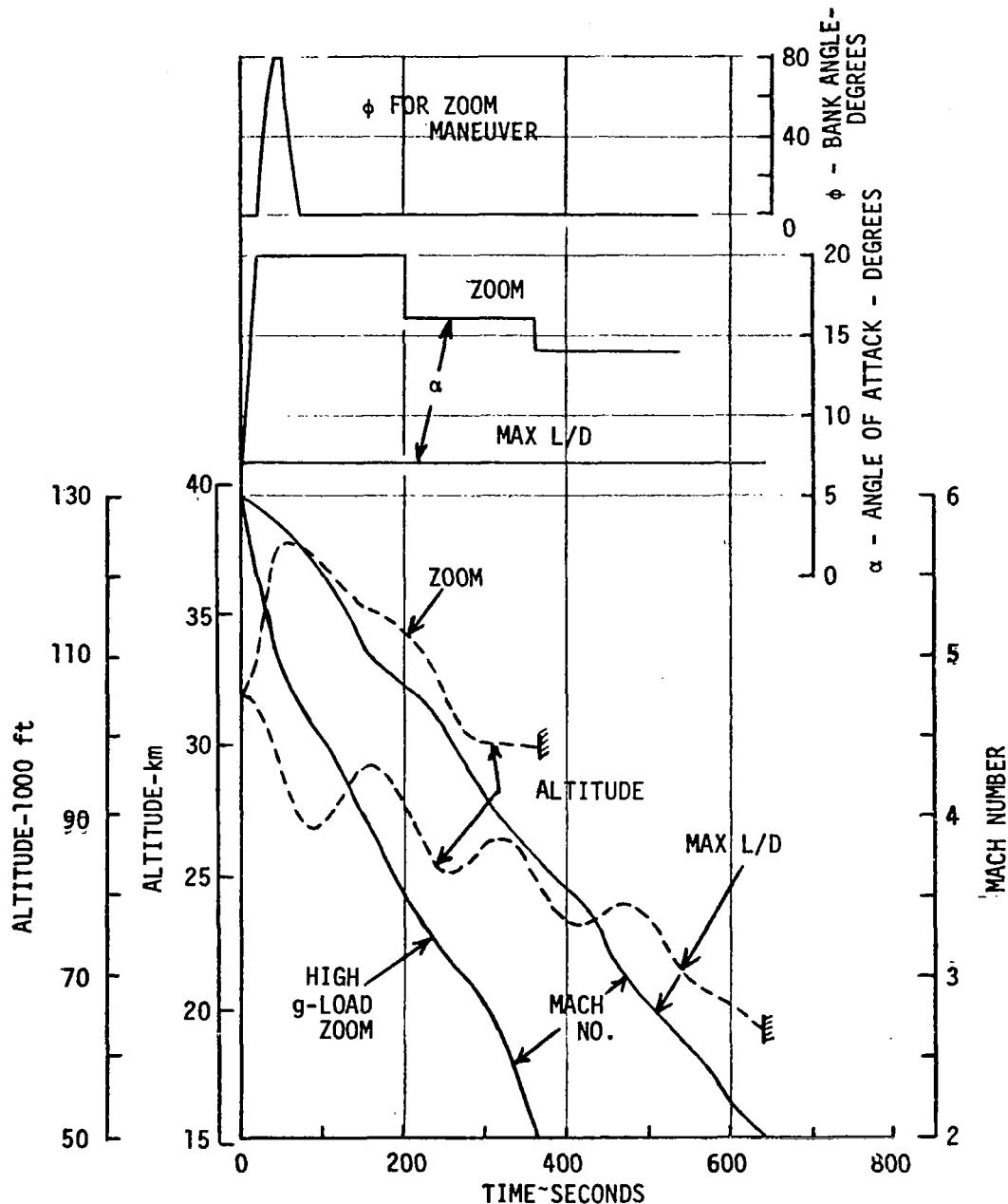


FIGURE 42  
MACH 6 DESCENT TRAJECTORIES

#### 5.4 FAIL-SAFE THERMOSTRUCTURAL CONCEPTS

The capability of the thermostructural concepts was evaluated on the basis of weight, considering maximum structural temperatures during abort, abort heating rates, and total heat load. An additional consideration was the structural cooling requirements during cruise, since some of the concepts could exceed the available heat sink capacity during cruise operation.

Selected combinations of upper and lower surface concepts were devised to provide visibility into the potential of each thermostructural concept (from the standpoint of overall aircraft integration). Figures 43, 44, and 45 present the baseline aircraft configuration total cooled area weight chargeable to the combination concepts (dry panel weight exclusive of structural support plus active cooling system weight). The average heat transfer to the coolant, for each upper/lower surface combination, is shown on the figures. An area-averaged approach was used in the comparison, where it was assumed that the typical upper and lower surface panels contributed 54.3 percent and 45.7 percent, respectively, of the total aircraft surface area. All configurations shown are capable of an abort from cruise, initiated 15 seconds after a total cooling system failure, without exceeding allowable material temperatures.

Mach 3 Concepts - As shown by Figure 43, the total weight of a silicone overcoat concept, when applied to both upper and lower surfaces, was found to be approximately the same as the baseline. In addition, average heat transfer to the coolant during cruise would be reduced to a level which can be absorbed by the hydrogen demanded by the aircraft engines. Weight of the polyimide overcoat concept was found to be considerably higher.

Mach 4.5 Concepts - The only combinations which resulted in a match between system heat load and available hydrogen heat sink utilize insulative heat shields or overcoats (Figure 44). An insulative heat shield on the lower surface, combined with a silicone overcoated upper surface could provide minimum cooling system heat load and would not be appreciably heavier than the baseline configuration. A bare Lockalloy lower surface combined with a silicone overcoated upper surface resulted in minimum total weight, but excessive cooling system heat load.

- o FAILURE RESPONSE TIME = 15 sec
- o TOTAL COOLED AREA =  $2985 \text{ m}^2$  ( $32,134 \text{ ft}^2$ )
  - $1621 \text{ m}^2$  ( $17,449 \text{ ft}^2$ ) UPPER SURFACE
  - $1364 \text{ m}^2$  ( $14,685 \text{ ft}^2$ ) LOWER SURFACE
- o PANEL SUPPORT STRUCTURE WEIGHT EXCLUDED



ACTIVELY COOLED PANEL WEIGHT  
ACTIVE COOLING SYSTEM WEIGHT  
HEAT TRANSFER TO COOLANT DURING CRUISE

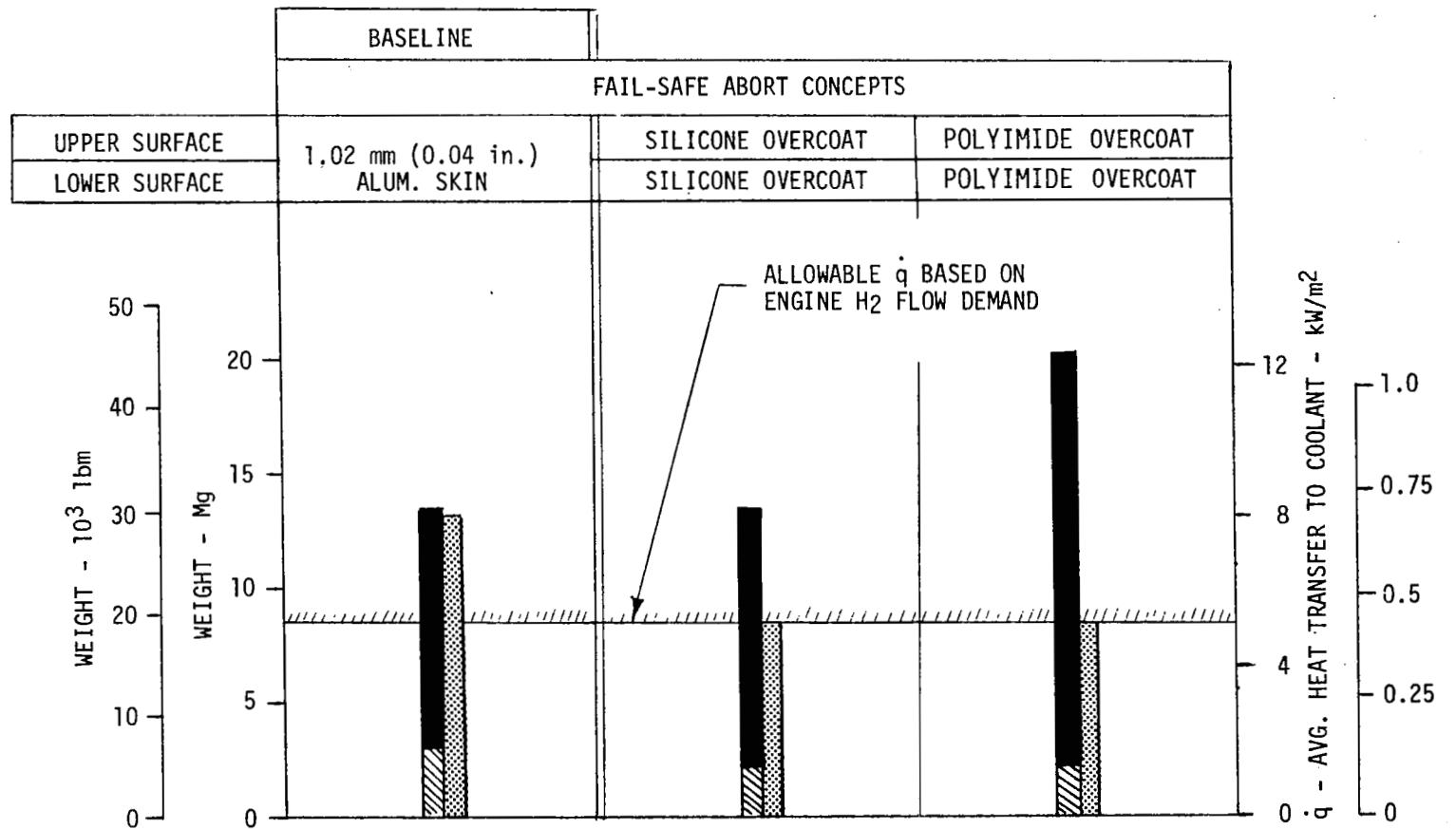


FIGURE 43  
MACH 3 THERMOSTRUCTURAL CONCEPTS

- o FAILURE RESPONSE TIME = 15 sec
- o TOTAL COOLED AREA = 2985 m<sup>2</sup> (32,134 ft<sup>2</sup>)
  - 1621 m<sup>2</sup> (17,449 ft<sup>2</sup>) UPPER SURFACE
  - 1364 m<sup>2</sup> (14,685 ft<sup>2</sup>) LOWER SURFACE
- o PANEL SUPPORT STRUCTURE WEIGHT EXCLUDED


 ACTIVELY COOLED PANEL WEIGHT  
 ACTIVE COOLING SYSTEM WEIGHT  
 HEAT TRANSFER TO COOLANT DURING CRUISE

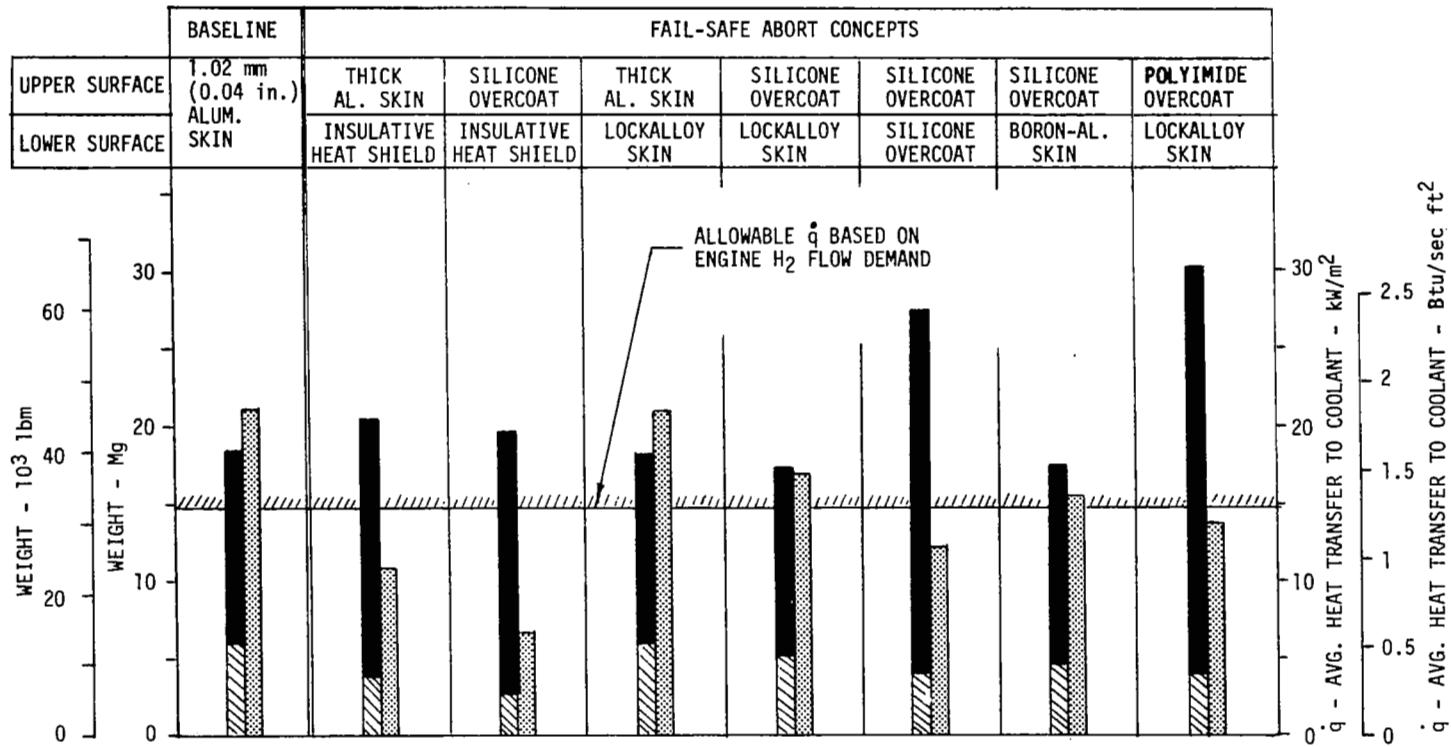


FIGURE 44  
MACH 4.5 THERMOSTRUCTURAL CONCEPTS

The combined concept of silicone overcoated upper surface with a boron-aluminum lower surface also resulted in an attractive total weight. However, the heat transfer to the coolant was again judged excessive.

Mach 6 Concepts - Figure 45 illustrates the actively cooled panel and cooling system weight chargeable to the investigated Mach 6 thermostructural concepts, as well as the average heat transfer to the coolant. As shown, utilization of any of the investigated thermostructural concepts on the lower surface other than an insulative heat shield results in excessive system heat loads. From both a total weight and absorbed heat transfer standpoint, the silicone overcoat upper surface combined with the insulative heat shield lower surface is the most attractive of the investigated concepts.

Lockalloy was not investigated as an upper surface material. However, the possibility of using Lockalloy on the upper surfaces in place of the overcoated aluminum skin should not be overlooked. It is expected that a combined Lockalloy upper surface - insulative heat shield lower surface would be weight competitive with the overcoated aluminum upper surface - heat shielded lower surface in addition to having an average level of absorbed heat transfer below the maximum allowable. The use of Lockalloy has the advantage of an easy inspection of the skin for crack propagation. In addition, the use of an all-shielded thermal protection system should be considered, again because this type of system can be readily inspected.

The possible use of Lockalloy skin provides another potential approach to the thermostructural design. The Lockalloy skin could be operated at an elevated temperature (in the 589 K (600°F) to 644 K (700°F) range) during cruise. This type of thermostructural design would still have available the temperature increase required to provide heat sink capacity during abort with its 799 K (800°F) allowable limit. The big advantage of letting the actively cooled panel operate at a higher temperature is that the panel cooling system could then be optimized to use a larger amount of the heat sink available from the hydrogen fuel. Design of the cooled panels to operate at the elevated temperature would require the cooling system to be designed to use silicone based coolant.

Thermostructural Concept Selection - A concept ranking process was performed to select candidate finalist concepts for each of the three abort Mach

- FAILURE RESPONSE TIME = 15 sec
- TOTAL COOLED AREA =  $2985 \text{ m}^2$  ( $32,134 \text{ ft}^2$ )
  - $1621 \text{ m}^2$  ( $17,449 \text{ ft}^2$ ) UPPER SURFACE
  - $1364 \text{ m}^2$  ( $14,685 \text{ ft}^2$ ) LOWER SURFACE
- PANEL SUPPORT STRUCTURE WEIGHT EXCLUDED


 ACTIVELY COOLED PANEL WEIGHT  
 ACTIVE COOLING SYSTEM WEIGHT  
 HEAT TRANSFER TO COOLANT DURING CRUISE

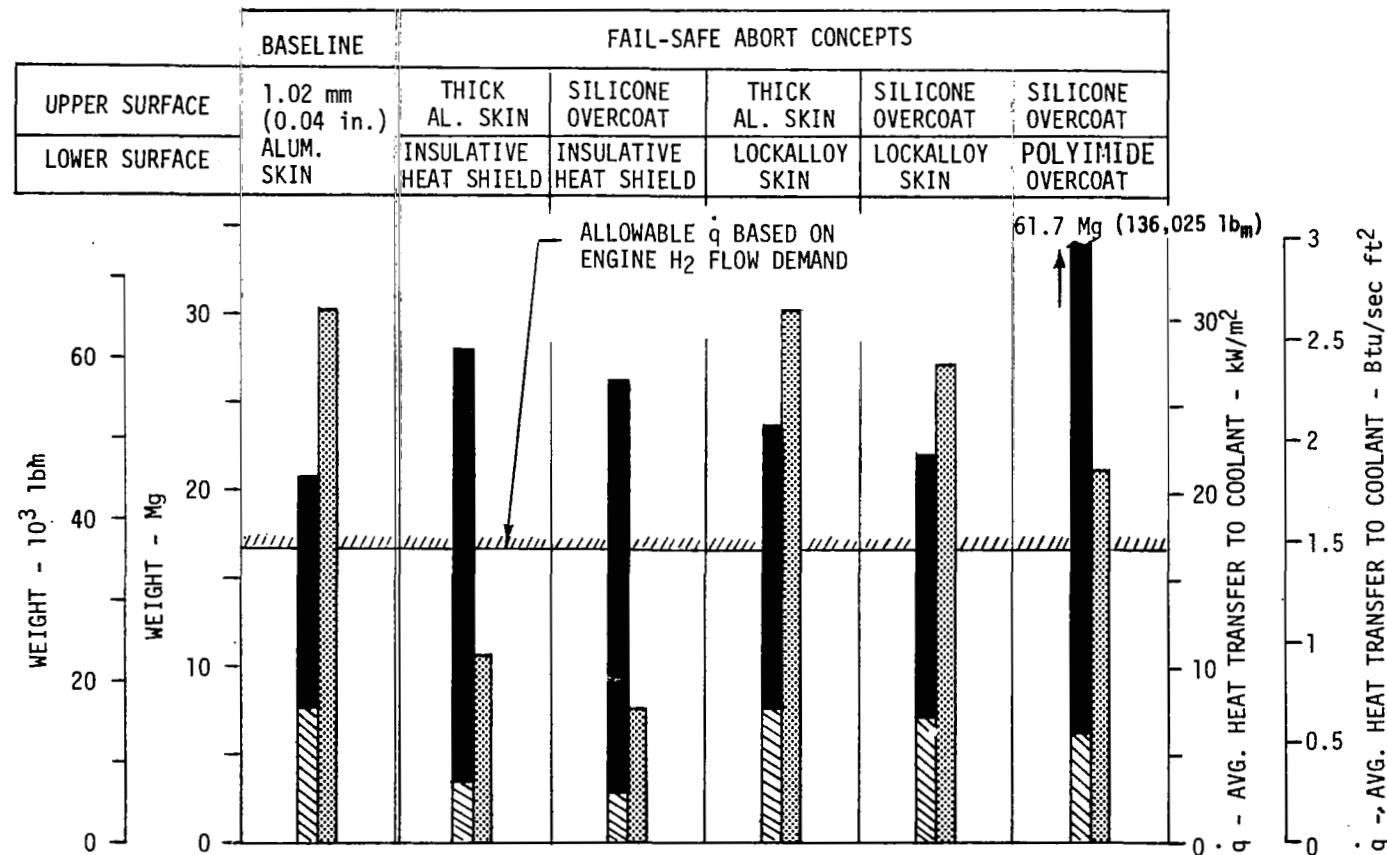


FIGURE 45  
MACH 6 THERMOSTRUCTURAL CONCEPTS

numbers. The concepts surviving the ranking/selection process were carried into a final systems integration and optimization phase.

Concept ranking required the establishment of a consistent and meaningful rating system if the end results were to be credible. Therefore, figures of merit were utilized in the concept ranking process which recognized the prime areas of importance. Wherever possible the merits of each design were quantified. This was easily done for weight and relative initial cost of the thermostructural concept-cooling system combinations. However, other figures of merit such as operating cost and reliability are functions of design complexity, damage resistance, damage tolerance, inspectability, maintainability and life, which could only be rated qualitatively. Thus, a concept ranking system allowing both quantitative and qualitative evaluations of the candidate thermostructural fail-safe concepts was established.

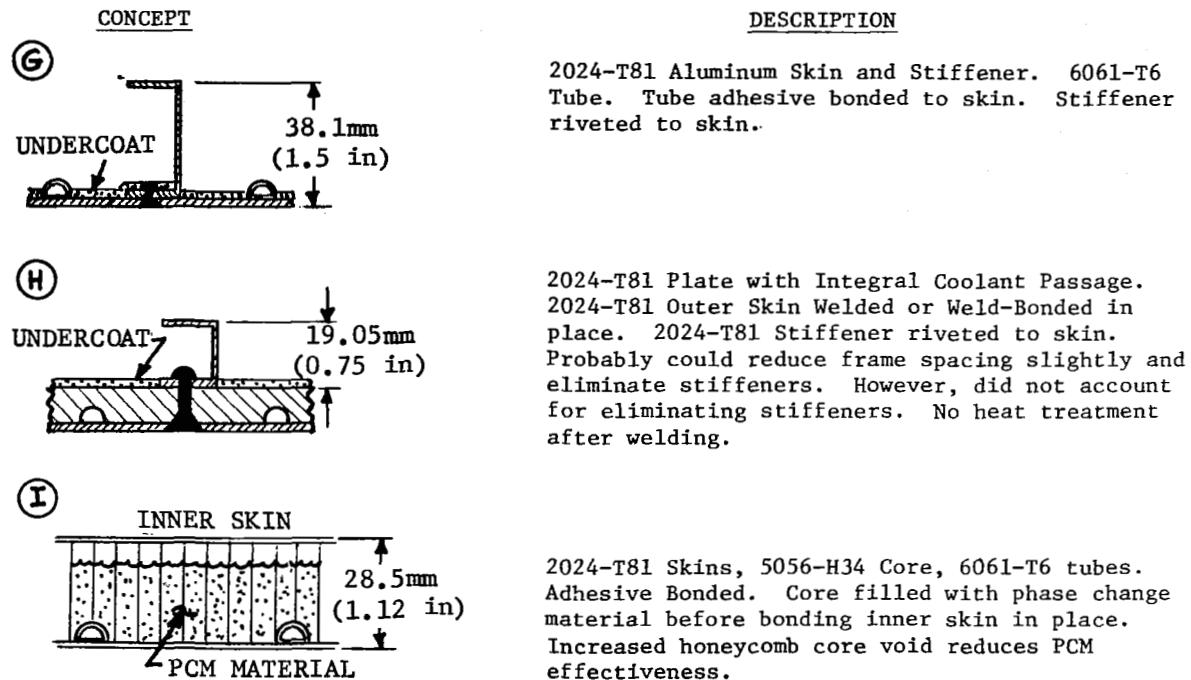
The structural concept applicable to each thermostructural approach were defined before proceeding with the ranking. Figure 46 shows the aircraft lower surface concepts configured for Mach 6 abort. Concepts for all other abort Mach numbers and surface locations were similar to one or another of the Mach 6 lower surface concepts. Some of the concepts considered will require considerable development; particularly the concepts using boron-aluminum, Lockalloy, surface coatings, or phase change materials. It was assumed that all required development could be successfully accomplished.

Each concept was rated in three major categories; operating cost, initial cost, and reliability. The considerations in the establishment of an operating cost figure-of-merit were total weight chargeable to a given concept (this included cooling system weight), damage resistance, inspectability, and maintainability. The initial cost figure-of-merit considered weight, and material and fabrication cost. The reliability figure-of-merit considered design tolerance to damage, design complexity and inspectability. These three figures-of-merit were then grouped into one overall figure-of-merit to provide a relative ranking of the concepts.

Complete details of this concept selection procedure are available in Reference (1). On the basis of the ranking of concepts, and the compatibility with available heat sink, the thermostructural concepts listed on Figure 47 were selected for further optimization. The finalist concepts are those presented in Section 4.4 herein.

<u>CONCEPT</u>	<u>DESCRIPTION</u>
<b>(A) BASELINE</b>	Honeycomb Panel, 2024-T81 Al. Skins, 5056-H34 Al. Core, 6061-T6 Al. Tubes, Adhesive Bonded.
<b>(B)</b>	2024-T81 Al. Plate With Integral Coolant Passage, 2024-T81 Outer Skin Welded or Weld-Bonded on Plate. No other Stiffening Rqd. No heat treatment after welding.
<b>(C)</b>	All Parts Boron-Aluminum. Metallurgical Joined.
<b>(D)</b>	Lockalloy Outer Skin (requires joining of relatively small size sheets), Lockalloy formed stiffeners, 6061-0 tube brazed to skin. No heat treatment after brazing.
<b>(E)</b>	2024-T81 Plate with Integral Coolant Passage, 2024-T81 Outer Skin Welded or Weld-Bonded in place. No other stiffening required. No heat treatment after welding.
<b>(F)</b>	Baseline Panel with Insulation and Preloaded Monolithic Heat Shield.

**FIGURE 46**  
**MACH 6 LOWER SURFACE STRUCTURAL CONCEPTS**



NOTE: STRUCTURAL CONCEPTS **A**, **B**, **C**, **D**, **E**, **F**, **G**, **H** AND **I** APPLICABLE TO THERMOSTRUCTURAL DESIGNS FOR ALL ABORT MACH NUMBERS AND AIRCRAFT SURFACES

**FIGURE 46 (Continued)**  
**MACH 6 LOWER SURFACE STRUCTURAL CONCEPTS**

In all cases, the insulative heat shield concepts were applied without gaps between the heat shield and insulation package or between the insulation and aluminum panel. This was done to eliminate degradation of thermal performance due to boundary layer leakage into, and through, the insulative system. Boundary layer leakage can increase the heat transfer rate to the insulated structure by significant amounts.

Strength analysis of the finalist thermostructural concepts were conducted to insure structural adequacy of the designs for both normal aircraft operation and for abort situations.

CRUISE/ABORT MACH NUMBER	TYPICAL AIRCRAFT SURFACE	THERMOSTRUCTURAL CONCEPT	BACK-UP STRUCTURE
3	UPPER	SILICONE OVERCOATED ALUMINUM SKIN	ALUMINUM HONEYCOMB
	LOWER	SILICONE OVERCOATED ALUMINUM SKIN	ALUMINUM HONEYCOMB
4.5	UPPER	SILICONE OVERCOATED ALUMINUM SKIN (PRIME)	ALUMINUM HONEYCOMB
		THICKENED ALUMINUM SKIN (SECONDARY)	ALUMINUM HONEYCOMB
	LOWER	TITANIUM HEAT SHIELDED ALUMINUM SKIN (PRIME)	ALUMINUM HONEYCOMB
		SILICONE OVERCOATED THICK ALUMINUM SKIN (SECONDARY)	SKIN/STRINGER
6	UPPER	SILICONE OVERCOATED ALUMINUM SKIN (PRIME)	ALUMINUM HONEYCOMB
		THICKENED ALUMINUM SKIN (SECONDARY)	ALUMINUM HONEYCOMB
	LOWER	INCONEL HEAT SHIELDED ALUMINUM SKIN (PRIME)	ALUMINUM HONEYCOMB
		(NO SECONDARY CONCEPT)	

FIGURE 47  
THERMOSTRUCTURAL CONCEPTS SELECTED FOR OPTIMIZATION

To maintain continuity, and to provide a basis for comparison with present actively cooled panel programs (see Footnote\*), typical design load requirements of the Reference (5) baseline aircraft were used to optimize the structural panels. These design loads are consistent with the footnoted studies. Figure 48 summarizes the design load requirements.

Eight of the nine finalist concepts were adhesively bonded aluminum honeycomb sandwich with coolant manifolds and dee-shaped tubes. The skins

\*NASA Contract NAS1-12919, "Design and Fabrication of an Actively Cooled Panel" and NAS1-13939, "Design and Fabrication of Radiative-Actively Cooled Structural Panel."

DESIGN LIMIT LOADS:  $n_x = +210.2 \text{ kN/m}$  ( $1200 \text{ lbf/in.}$ )

Pressure =  $+6.9 \text{ kPa}$  (1.0 psi)

ULTIMATE LOADS = 1.5 times design limit loads

FATIGUE AND FRACTURE MECHANICS DESIGN CRITERIA:

- a) Cyclic loading of  $+210.2 \text{ kN/m}$  ( $1200 \text{ lbf/in.}$ ) (stress ratio = -1)
- b) 5000 cycles at limit design temperatures (20,000 cycles with a scatter factor of 4)
- c) Constant uniform lateral pressure load,  $6.9 \text{kPa}$  (1.0 psi)
- d) Cracks will not grow through the wall thickness of coolant passages in 20,000 cycles.

**FIGURE 48**  
**STRUCTURAL PANEL LOAD REQUIREMENTS**

were 2024-T81 and the tubes 6061-T6 aluminum. A two foot intermediate frame spacing was used in the analysis. The coolant pressure in the tubes was assumed to be  $690 \text{ K Pa}$  (100 psi). The honeycomb core was  $49.7 \text{ kg/m}^3$  ( $3.1 \text{ lbm/ft}^3$ ) material. Utilizing geometric inputs provided by thermal analysis of panel requirements the honeycomb structural panels were optimized. The resulting honeycomb panel structural weights are summarized in Figure 49.

One finalist thermostructural concept (the Mach 4.5 aircraft lower surface candidate with silicone overcoated aluminum skin) was assumed to be of skin/stringer construction. This concept was evaluated for the same load requirements as the honeycomb structure. Results showed that a skin/stringer panel would be about the same weight as a comparable honeycomb panel. The actual panel concept construction would have a thick, 3.56 mm (0.14 in.), skin due to thermal requirements. The weight of this concept was  $15.48 \text{ kg/m}^2$  ( $3.17 \text{ lbm/ft}^2$ ), about 22 percent heavier than an equivalent honeycomb panel. Thus, the skin-stringer concept was judged to be more than adequate from a structural viewpoint if designed to meet the thermal requirements.

The structural weights derived from these analyses are the basis for the structural weights presented in Section 4.

MACH NO.	AIRCRAFT SURFACE	$h'$ cm (in)	$t_1$ mm (in)	UNIT STRUCTURAL WEIGHT $\text{kg/m}^2 (1\text{lb}_m/\text{ft}^2)$			TOTAL PANEL (DRY)
				OUTER SKIN + TUBES	CORE + INNER SKIN	$\Delta$	
3	(U)Upper	3.15 (0.24)	1.07 (0.042)	3.213 (0.658)	4.453 (0.912)	4.98 (1.02)	12.65 (2.59)
	(L)Lower	3.15 (1.24)	1.02 (0.040)	3.384 (0.693)	4.282 (0.877)	4.98 (1.02)	12.65 (2.59)
4.5	(U)	3.0 (1.18)	1.02 (0.040)	3.447 (0.706)	4.218 (0.864)	4.98 (1.02)	12.65 (2.59)
	(L) $\Delta$ Insulated	3.15 (1.24)	1.07 (0.042)	3.232 (0.662)	4.433 (0.908)	4.98 (1.02)	12.65 (2.59)
6	(U)	3.0 (1.18)	0.99 (0.039)	3.520 (0.721)	4.145 (0.849)	4.98 (1.02)	12.65 (2.59)
	(L) $\Delta$ Insulated	3.15 (1.24)	1.07 (0.042)	3.232 (0.662)	4.433 (0.908)	4.98 (1.02)	12.65 (2.59)

NOTES:  $\Delta$  1 OVERCOAT, HEAT SHIELD AND INSULATION WEIGHTS MUST BE ADDED TO STRUCTURAL WEIGHT INCREMENTS.

$\Delta$  2 TYPICAL VALUES OF PANEL GEOMETRY, WHERE:

$h'$  = DISTANCE BETWEEN CENTROIDS OF OUTER AND INNER SKIN.

$t_1$  = THICKNESS OF INNER SKIN

$\Delta$  3 WEIGHT INCREMENT FOR MANIFOLDS, PANEL ATTACHMENTS, AND NONOPTIMUMS.

$\Delta$  4 ADD  $2.43 \text{ kg/m}^2 (0.498 \text{ lb}_m/\text{ft}^2)$  FOR TITANIUM HEAT SHIELD (CORRUGATION STIFFENED BEADED SKIN) AND SUPPORTS.

$\Delta$  5 ADD  $4.35 \text{ kg/m}^2 (0.89 \text{ lb}_m/\text{ft}^2)$  FOR RENE' 41 HEAT SHIELD (CORRUGATION STIFFENED BEADED SKIN) AND SUPPORTS.

FIGURE 49  
HONEYCOMB PANEL STRUCTURAL WEIGHT SUMMARY

## 6. CONCLUSIONS

The potentials of a "Fail-Safe Abort System" for use in actively cooled, hydrogen fueled, supersonic and hypersonic aircraft were examined. This concept depends on three basic elements:

- o Detection of cooling system malfunctions, including individual actively cooled structural panels; with reliable and responsive sensors.
- o An abort descent trajectory which minimizes the aircraft descent heat load.
- o Fail-safe thermostructural concepts which minimize the increase in structural temperatures during abort at minimum weight.

Conclusions relative to these elements follow:

- a. Detecting Cooling System Failure is State-of-the-Art - Basic cooling system malfunctions, or failure, can be detected by the use of conventional instrumentation such as pressure transducers, flow meters, thermistors and thermocouples, and liquid level indicators for measuring system parameters.
- b. Detecting Loss of Cooling in the Airframe Panels is Feasible - Failure (loss of cooling) of individual actively cooled structural panels can be detected by temperature sensitive elements attached to the aluminum skins of the cooled panels. The temperature sensor response times appear to be such that no significant weight penalty will be experienced due to delay between failure and failure detection. The temperature sensing elements required for detection of individual panel loss of cooling, and the associated system components, are relatively complex and require development. The potential benefits of these sensors appear to justify the development risk.
- c. Abort Trajectories Significantly Reduce Heat Load - Abort trajectories using constant g-load pull-up maneuvers result in descent heat loads sufficiently low that abort heating does not result in significant thermostructural weight penalties.
- d. No Weight Penalty Required for Fail-Safe Features - No thermostructural weight penalty due to abort heating is incurred for Mach 3 or Mach 4.5 cruise aircraft. Design of the thermostructural system to provide a match between absorbed heat and hydrogen fuel heat sink capacity provides a thermostructural design capable of abort descent. The weight penalty due to Mach 6 abort is offset by reductions in weight of the active cooling system. The

"Fail-Safe" concept, compared to an unprotected aluminum structure aircraft carrying adequate hydrogen to meet all cooling requirements, will result in a reduction of the combined structural/cooling system/required heat sink weight.

e. Aircraft is Reusable After Abort - Structural temperatures during abort from Mach 3 and Mach 4.5 can be kept below 450 K (350°F), the practical limit for reuse of the aluminum structure. The thermostructural design, at these Mach numbers, is dictated by cruise Mach number considerations. The maximum structural temperatures encountered during Mach 6 abort are a function of the type of failure and the time required to sense the failure. Basic cooling system malfunction, or failure of a surface panel to receive adequate cooling will result in maximum abort temperature levels on the order of 443 K (337°F) and 465 K (377°F), for the typical upper surface and lower surface structural concepts studied.

THE BASIC CONCEPT OF "FAIL-SAFE ABORT SYSTEMS" IS COMPLETELY FEASIBLE THROUGHOUT THE MACH 3 TO MACH 6 SPEED RANGE.

In addition to the primary conclusions stated above, a comparison of results of this study with those of Reference (7) resulted in several pertinent observations. These are:

- o The addition of "Fail-Safe" capability to an actively cooled Mach 6 cruise transport design results in total system unit area weights approximately equal to the total system unit area weights for a heat shielded design with redundant cooling systems.
- o A marriage of a redundant cooling system approach with the "Fail-Safe Abort" concept appears possible with minimal weight impact. An integrated system of this type would have outstanding overall safety capability.
- o The structural heat protection required to provide the "Fail-Safe Abort" capability allows a reduction in aircraft heat loads to a point where normal engine fuel flow demands can provide more than adequate heat sink.

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